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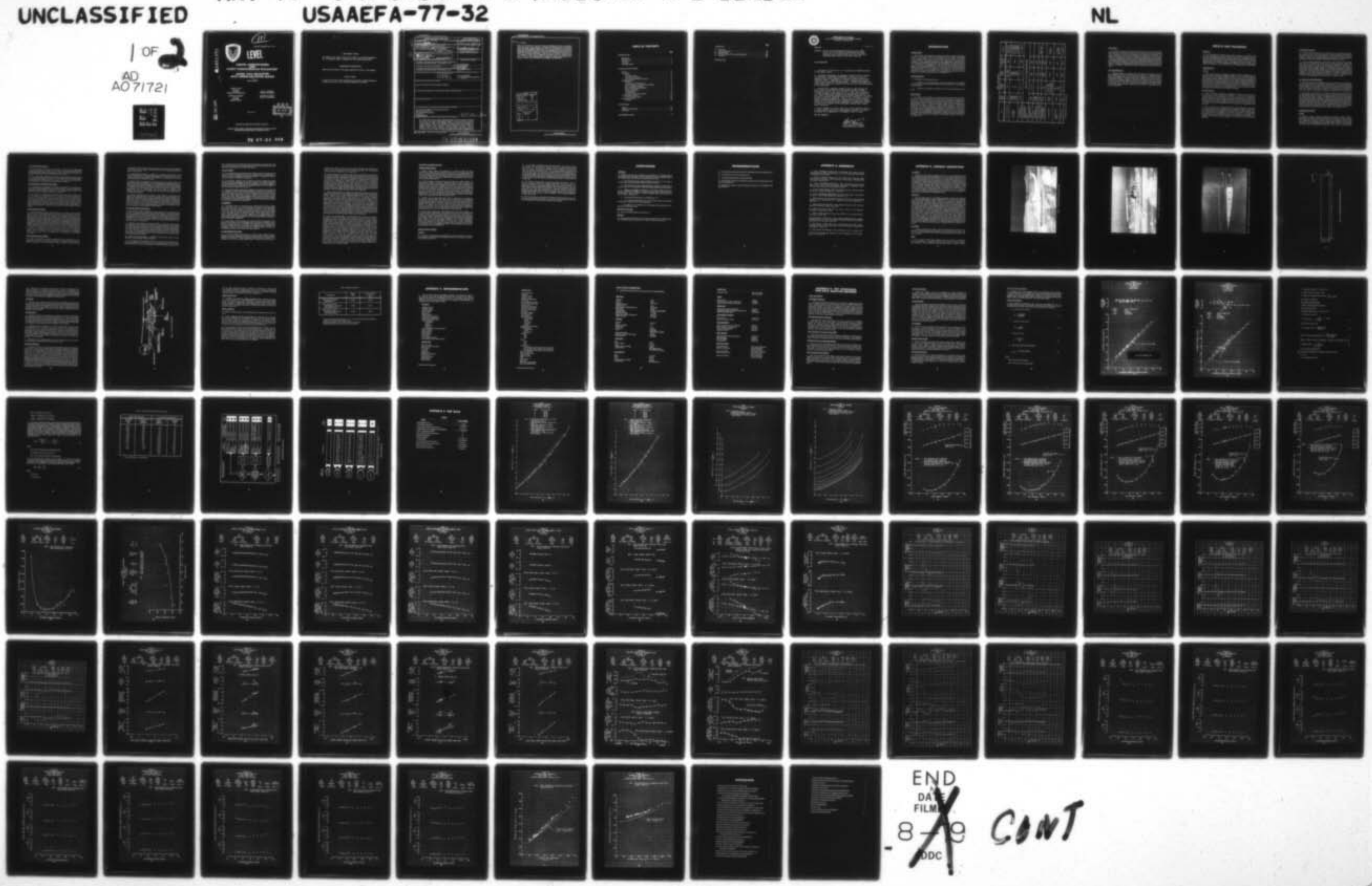

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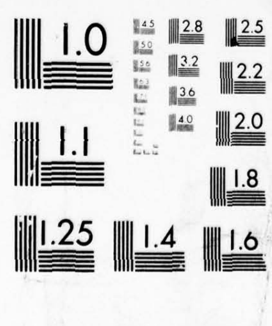

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LIMITED AIRWORTHINESS AND FLIGHT CHARACTERISTICS EVALUATION

MODEL 214A HELICOPTER WITH FIBERGLASS Main ROTOR BLADES

FINAL REPORT

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MAY 1979

UNITED STATES ARMY AVIATION ENGINEERING FLIGHT ACTIVITY
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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The United States Army Aviation Engineering Flight Activity conducted a limited Airworthiness and Flight Characteristics evaluation of the Model 214A helicopter to determine the effect of the fiberglass main rotor blades on performance and handling qualities. Testing was conducted at the Bell Helicopter Textron Flight Test Facility, Arlington, Texas (630-foot elevation), Bishop, California (4120-foot, 9980-foot, and 11,750-foot elevation), and Edwards Air Force Base, California (2302-foot elevation) during the period 29 August 1978 through 13 October 1978.

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20. Abstract

Thirty-six test flights for a total of 20.4 productive flight hours were accomplished. Within the scope of this evaluation, the Model 214A helicopter with the fiberglass main rotor blades exhibited slightly improved hover and level flight performance at higher gross weight-altitude combinations. The handling qualities remain essentially unchanged from those exhibited by the Model 214A helicopter with standard rotor blades. The enhancing characteristic noted was the low level-flight vibration levels. The deficiency noted was the inadequate directional control at airspeeds greater than 15 KTAS in right sideward flight at high gross weight, density altitude combinations, which is not attributable to the fiberglass rotor installation.

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DEPARTMENT OF THE ARMY
HQ, US ARMY AVIATION RESEARCH AND DEVELOPMENT COMMAND
P O BOX 209, ST. LOUIS, MO 63166

8 June 1979

DRDAV-EQ

SUBJECT: US Army Aviation Engineering Flight Activity (AEFA) Project No. 77-32, Limited Airworthiness and Flight Characteristics Evaluation Model 214A Helicopter With Fiberglass Rotor Blades, Final Report, May 1979

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1. The purpose of this letter is to establish the AVRADCOM position on the subject report.

2. The general conclusions of paragraph 45 appear correct; however, the performance improvements documented in this report were not expected in that rotor geometry is identical between the metal and fiberglass blades. Improvements in surface smoothness and contour control could account for the improvement. A similar improvement was noted during contractor testing for the FAA for the civil certification of the same blade.

3. The deficiency regarding sideward flight at high altitude and heavy weight (on the order of 13,600 lbs and 10,000 ft density altitude) is common to the basic aircraft. Under these conditions, a 15 KTAS sideward flight capability is relative good, compared to most current helicopters designed to MIL-H-8501A standards, and there has been no reported complaints from the user; therefore, no action is planned reference paragraph 50. Since the user no longer procures changes to technical publications through the US Army, this Command cannot take the specific action recommended by paragraph 52; however, the point can be handled by Bell Helicopter Textron (BHT) through service bulletins.

4. Due to changes in the GOI, production of the fiberglass blade is no longer planned under any US Army auspices; therefore, no action concerning the recommendation in paragraph 53 will be taken.

FOR THE COMMANDER:

CHARLES C. CRAWFORD, JR.
Chief, Systems Development
and Qualification Division

INTRODUCTION

BACKGROUND

1. The US Army Troop Support and Aviation Readiness Command (TSARCOM) awarded a development contract to Bell Helicopter Textron (BHT) in September 1977, to design, fabricate, and test a fiberglass main rotor blade for the Model 214A/C helicopter. The design objectives of the program were to improve safety, reliability, and maintainability characteristics. The resulting BHT design includes the use of composite material construction, a swept tip planform, and the same airfoil as the 214 metal blade. In September 1977, the US Army Aviation Research and Development Command (AVRADCOM) directed the US Army Aviation Engineering Flight Activity (USAAEFA) to conduct an Airworthiness and Flight Characteristics Evaluation (A&FC) of the fiberglass main rotor blades for the Model 214A/C helicopter (ref 1, app A). The Test Plan for the conduct of the evaluation (ref 2) was prepared by USAAEFA and approved by AVRADCOM (ref 3).

TEST OBJECTIVES

2. The objectives of this evaluation were:

a. Confirm that the airworthiness and flight characteristics of the Model 214A helicopter are not adversely affected by the incorporation of the fiberglass main rotor blade.

b. Determine the degree to which the test item meets specification requirements.

DESCRIPTION

3. The Model 214A helicopter is a utility helicopter derived from the UH-1 series. It is being procured by the United States Government for the Imperial Government of Iran from BHT. The test aircraft has a two-bladed, fiberglass teetering main rotor and is powered by one Lycoming T5508D turboshaft engine having a thermodynamic rating of 2930 shaft horsepower (shp) at sea-level, standard day conditions. The engine is derated to a torque rating equivalent to 2250 shp at 100 percent rotor speed; however, the aircraft drive train is limited to 2050 shp for five minutes and 1845 shp continuous. Provisions are included for a crew of two and various payload configurations that include space for 14 troops. The mission of the aircraft is transportation of personnel, special teams or crews, equipment and supplies; medical evacuation; and instrument training. A further description of the Model 214A helicopter is contained in the Operator's Manual (ref 4, app A) and in appendix B.

Table 1. Test Conditions.

Test	Gross Weight (lb)	Calibrated Airspeed (lb)	Density Altitude (ft)	Longitudinal Center-of-Gravity	Remarks
Hover performance	N/A	zero	4920, 9980 and 12,900	MID	10 and 100 ft skid height
Level flight performance	10,380 10,320 13,280 13,340 13,240	40 to V_H[1]	6460 12,040 8,440 12,540 16,060	FWD	$C_T = 0.00440$ $C_T = 0.00518$ $C_T = 0.00595$ $C_T = 0.00680$ $C_T = 0.00761$
Autorotational descent performance	13,280	55 to 112	9720	FWD	$C_T = 0.00620$
Control positions in trimmed forward flight	SAME AS LEVEL FLIGHT PERFORMANCE				
Static longitudinal stability	13,800	77 and 125	5000	MID	
Static lateral-directional stability	13,800 13,500	37	5000	MID	
Maneuvering stability	13,500	126	5000	AFT	Right turns
Dynamic stability and controllability	13,500	zero 124	2000 6500	AFT AFT	
Low speed flight	13,640 13,500	0 to 35 left 0 to 15 right 0 to 30 forward 0 to 30 rearward	10,000	FWD	Skid height 10 feet
Simulated engine failures	10,000	40 46 116 140	7,000	AFT	Level flight and climbs
Vibration levels	10,000	40 to V_H[1]	10,000	FWD	Level flight

[1] V_H: Maximum airspeed for level flight.

TEST SCOPE

4. The A&FC testing of the Model 214A helicopter was conducted at the BHT Flight Test facility, Arlington, Texas (630-foot elevation), Bishop, California (4120-foot, 9980-foot, and 11,750-foot elevations), and Edwards Air Force Base, California (2302-foot elevations), from 29 August 1978 through 13 October 1978. Thirty-six test flights for a total of 20.4 productive flight hours were accomplished at the test conditions shown in table 1. The test aircraft, BHT serial number 27001, was provided, instrumented, and maintained by BHT. Flight limitations contained in the Airworthiness Release (ref 5, app A) and the Operator's Manual were observed during the evaluation. Handling qualities and vibration levels were evaluated with respect to the applicable requirements of Military Specification MIL-H-8501A (ref 6, app A).

TEST METHODOLOGY

5. Established flight test techniques and data reduction procedures were used (refs 7 through 9, app A) and are described in appendix D. A Handling Qualities Rating Scale (HQRS) (fig 1, app D) was used to augment pilot comments relative to handling qualities. A Vibration Rating Scale (VRS) (fig. 2, app D) was used to augment pilot comments relative to vibration. The flight test data were obtained from test instrumentation displayed on the pilot and copilot panels and recorded on magnectic tape. Final data reduction was accomplished at the BHT facilities at Hurst and Arlington, Texas. A detailed list of the test instrumentation is contained in appendix C.

RESULTS AND DISCUSSION

GENERAL

6. An Airworthiness and Flight Characteristics evaluation of the Model 214A helicopter with fiberglass main rotor blades was performed to determine any differences on performance and handling qualities caused by the fiberglass main rotor. The enhancing characteristic was noted during the evaluation: the low level flight vibration levels. The deficiency noted, the inadequate directional control at airspeeds greater than 15 KTAS in right sideward flight, at high gross weight, density altitude combinations, is not attributable to the fiberglass main rotor installation.

PERFORMANCE

General

7. Hover, level flight, and autorotational descent performance testing were conducted at Bishop, California (4120-foot, 9980-foot, and 11,750-foot elevation) and Edwards Air Force Base, California (2302-foot elevation). The performance data obtained with the fiberglass main rotor were compared with the standard blade data presented in reference 10, appendix A. Minor external configuration differences between the two test aircraft are discussed in appendix B. No attempt was made to apply corrections for these differences. The hover and level flight performance of the Model 214A helicopter with the fiberglass main rotor is improved at higher values of thrust coefficient (C_T) and slightly degraded at lower values of C_T. Autorotational descent performance was essentially unchanged.

Hover Performance

8. Hover performance testing was accomplished at Bishop, Coyote Flats, and Piute Mountain, California (4120-foot, 9980-foot, and 11,750-foot, respectively) at skid heights of 10 and 100 feet. The aircraft was tethered to the ground to maintain the proper skid height while incrementally varying engine power and rotor speed. A load cell was used to measure tension in the tether cable, and this tension was added to the aircraft gross weight to determine main rotor thrust. Figures 1 and 2, appendix E, present the nondimensional hover performance data gathered during these tests.

9. Hover performance at both 10 and 100 feet is slightly improved at high values of thrust coefficient (C_T) by installation of the fiberglass rotor blades. At low values of C_T (below 0.0042 IGE, 0.0052 OGE) the performance is slightly degraded. The out-of-ground-effect hover capability is increased by 175 pounds to 13,975 pounds, at a pressure altitude of 5000 feet, and a 35°C temperature, (the Model 214A helicopter metal blade guarantee conditions), the 10-foot hover capability is increased by 459 pounds, to a gross weight of 15,332 pounds at the same ambient conditions.

Level Flight Performance

10. The level flight performance of the Model 214A with fiberglass rotor blades was evaluated at five thrust coefficients (C_T). Thrust coefficients from 0.0044 to 0.00761 were flown at zero sideslip. Altitude was increased between data points to maintain a constant ratio of gross weight to air density (w/ρ) as fuel was consumed. The results of these tests are presented in figures 3 through 9, appendix E.

11. The level flight performance of the Model 214A helicopter with fiberglass rotor blades is not significantly changed from the performance of the Model 214C with metal blades (ref 11, app A). At values of C_T of 0.00595 and greater, the performance is slightly improved with the fiberglass blades and at lower values of C_T the performance is slightly degraded. At the Model 214A guarantee conditions of 5000 feet density altitude, standard day, at 13,000 pounds gross weight, the Model 214A with metal blades had a maximum level flight true airspeed (V_H) capability of 157 knots, true airspeed (KTAS) at maximum continuous power (1845 shp). With fiberglass blades, this airspeed was reduced to 155 KTAS, which exceeds the 144 KTAS guarantee and the never-exceed velocity, (V_{NE}).

Autorotational Descent Performance

12. Autorotational descent performance tests were conducted at 13,280 pounds, forward center of gravity, and 9720 feet density altitude. A series of autorotational descents (needles split) were conducted at constant rotor speed while incrementally varying airspeed. A second series of descents was conducted at constant airspeed (near the minimum-rate-of-descent airspeed) while incrementally varying rotor speed. The data from these tests are presented in figures 10 and 11, appendix E.

13. The airspeed for minimum rate of descent ($V_{min\ R/D}$) was 70 KCAS and is not significantly changed from the 71 KCAS $V_{min\ R/D}$ of the Model 214A with metal rotor blades (fig. 10, app E). The minimum rate of descent at this airspeed was 1880 feet-per-minute compared to 2000 feet-per-minute with metal blades. The best glide airspeed (*ie.*, the airspeed for best range in autorotation) was 95 knots calibrated airspeed (KCAS) and is not significantly changed from the 94 KCAS of the Model 214A with the metal rotor blades. The rotor speed for minimum rate of descent occurred below the power OFF continuous lower limit of 285 rpm. The autorotational descent performance of the Model 214A helicopter with fiberglass main rotor blades is essentially unchanged from that of the standard blades.

HANDLING QUALITIES

General

14. Handling qualities testing was conducted at Arlington, Texas (630-foot elevation), Coyote Flats, California (9980-foot elevation), and Edwards Air Force Base, California (2302-foot elevation). Testing was conducted with SCAS-ON and attitude hold disengaged, unless otherwise stated. Within the scope of this evaluation, the handling qualities of the Model 214A helicopter with fiberglass main rotor blades is essentially unchanged from those with the standard blades.

Control System Characteristics

15. The mechanical characteristics of the flight control system were qualitatively evaluated throughout the conduct of these tests. The breakout and friction forces in the longitudinal, lateral, and directional control systems remain essentially unchanged from previous Model 214A helicopters (ref 10, app A).

16. The objectionable collective control forces found previously (ref 10, app A) were not present during this evaluation, but the difference was not attributable to the fiberglass blades. Within the scope of this test, the control sytem characteristics were unchanged by the installation of the fiberglass rotor blades.

Control Positions in Trimmed Forward Flight

17. Control positions in trimmed zero-sideslip forward flight were evaluated from 40 to 149 KCAS with SCAS ON and attitude hold OFF. Tests were conducted in conjunction with level flight performance at the conditions listed in table 1. Test results are presented in figures 12 through 15, appendix E.

18. The variation of longitudinal control trim position with airspeed during trimmed level flight was essentially linear with increasing forward cyclic displacement required for increasing airspeed. Lateral and directional control trim position variation with airspeed change were minimal, with less than 1.5 inch of lateral or directional control motion required over the entire test airspeed range. Trim pitch attitude varied approximately 5 degrees nose down as airspeed increased from 40 to 149 KCAS. All control margins were adequate. The control position characteristics in trimmed forward flight are satisfactory and similar to previous Model 214A helicopters.

Static Longitudinal Stability

19. Static longitudinal stability characteristics were evaluated at the conditions listed in table 1. Longitudinal stability characteristics were evaluated by first trimming the aircraft at the desired trim airspeed. While holding collective fixed, the helicopter was displaced from the trim airspeed and stabilized at airspeeds greater and less than the trim airspeed. Data were recorded at each stabilized airspeed and are presented in figure 16, appendix E. The variation of longitudinal control position with airspeed indicated positive static stability (aft control displacement for slower airspeeds) under all test conditions. The longitudinal control position gradients were shallow, which gave insignificant displacement cues at the high airspeed trim point (124 KCAS). Displacement cues were more apparent at the low airspeed trim point (77 KCAS). The longitudinal control forces associated with cyclic displacements around both trim airspeed did not provide significant pilot force cues for airspeed changes. These undesireable static longitudinal characteristics were previously reported (ref 10, app A) and are unchanged with the installation of the fiberglass rotor blades.

Static Lateral-Directional Stability

20. Static lateral-directional stability characteristics were evaluated at the conditions shown in table 1. Tests were conducted by trimming the aircraft in zero-sideslip level flight at the desired airspeed. With the collective control fixed and

maintaining a steady heading and the trim airspeed, the aircraft was then stabilized at incremental sideslip angles on both sides of trim. Test results are presented in figure 17, appendix E.

21. Static directional stability, as indicated by the variation of directional control position with sideslip, was positive (right pedal with left sideslip) at both test airspeeds. Directional control variation with sideslip was essentially linear at all airspeeds and increased as airspeed increased.

22. Dihedral effect, as indicated by the variation of lateral control position with sideslip, was positive (left lateral control in left sideslip) and essentially linear at the slower trim airspeed. In left sideslips at a trim airspeed of 125 KCAS, the dihedral effect became increasingly more positive as sideslip was increased. In right sideslips at 125 KCAS, the dihedral effect became neutral as sideslip was increased. These nonlinearities were not objectionable and did not cause any aircraft control difficulty since they occur at sideslip angles greater than are normally incurred during operational flying.

23. Side-force characteristics, as indicated by the variation of bank angle with sideslip, were essentially linear at both test airspeeds and strongly positive at the higher trim airspeed. At the trim airspeed of 37 KCAS, the side-force characteristics were weak. At the slower trim airspeed, pilot cues with respect to sideslip were minimal; however, the sideslip envelope is large (greater than 40 degrees at airspeeds below 65 KCAS) and exact control of sideslip is not operationally critical. Within the scope of this test, the static lateral-directional stability characteristics are satisfactory and similar to previous Model 214A helicopters.

Maneuvering Stability Characteristics

24. Maneuvering stability characteristics were evaluated at the conditions listed in table 1 by trimming the aircraft in zero-sideslip level flight at the desired airspeed and then establishing a steady-state banked turn. The collective was held fixed at the level flight trim setting, and constant airspeed was maintained during the turn by varying altitude as required. Data were recorded through a 2000-foot altitude band, ±1000 feet from the trim test altitude. Results of the maneuvering stability tests are presented in figure 18, appendix E.

25. Stick-fixed and stick-free maneuvering stability, as indicated by the variation of longitudinal control position and force with normal acceleration were positive (aft control movement and pull force with increasing load factor) at the test airspeed and aircraft loading at load factors up to approximately 1.36g. The aircraft exhibited neutral stick-fixed maneuvering stability at normal acceleration values greater than 1.36g. This characteristic was previously identified (ref 10, app A) as being attributable to the pitch SCAS actuator reaching full extension and was unchanged by the fiberglass blade installation.

26. During maneuvering flight at 126 KCAS, precise lateral control became difficult at roll attitudes greater than 40 degrees.

27. At the airspeed tested, stabilized turns to the right generally required increasing right cyclic displacement from the trim point up to approximately 1.23g's. At load factors greater than 1.23g's, the requirement for right lateral cyclic decreased up to

the maximum load factor tested. The maneuvering stability characteristics of the Model 214A helicopter with fiberglass main rotor blades were essentially unchanged from those of the standard rotor, and are satisfactory.

Dynamic Stability

28. The longitudinal and lateral-directional dynamic stability of the Model 214A helicopter with the fiberglass main rotor was evaluated at the conditions listed in table 1, using the test techniques described in appendix D. Representative data are presented in figures 19 through 24, appendix E.

29. The short-term response was well damped with SCAS engaged and lightly damped with SCAS desengaged. The primary response to external gust inputs was a mild lateral-directional oscillation that was well-damped and easily controlled by the pilot; however, the aircraft did not consistently return to the trim flight conditon, as previously reported (ref 10, app A).

30. Long period dynamic response was evaluated at 112 KCAS with the SCAS ON by slowing the aircraft 10 knots indicated airspeed (KIAS) below trim airspeed and then returning the cyclic control to the trim position. Aircraft response was either neutrally or lightly damped with a period of oscillation greater than 20 seconds. Mission tasks which require precise airspeed control, such as IFR flight, required minimal pilot effort to accomplish (HQRS 3). The dynamic stability characteristics of the Model 214A helicopter with fiberglass main rotor blades remain unchanged from the standard blades and are satisfactory.

Controllability

31. Controllability tests were conducted at the conditions shown in table 1. Single-axis control step inputs of several magnitudes were applied to the longitudinal, lateral, and directional controls, using mechanical fixtures to obtain the desired control input size. The step inputs were made while other controls were maintained at the trim position and the subsequent aircraft angular displacement (control power), maximum angular rate (control response), and maximum angular acceleration (control sensitivity) were recorded. Results of the controllability tests, accomplished during steady trimmed flight in a hover and forward flight, are presented in figures 25 through 29, appendix E.

32. Control power, control response, and control sensitivity characteristics of the Model 214A helicopter with the fiberglass main rotor blades are similar to those observed during previous tests of the Model 214A helicopter with a standard main rotor blade. Control harmony was satisfactory in all flight regimes; no adverse cross coupling or tendency to overcontrol was noted. Within the scope of the test, the controllability characteristics are unchanged and satisfactory.

Low-Speed Flight Characteristics

33. The low-speed flight characteristics of the Model 214A helicopter with the fiberglass main rotor blades were evaluated at the conditions listed in table 1. Testing was performed to 30 KTAS rearward, 30 KTAS forward, 35 KTAS in left sideward, and 15 KTAS in right sideward flight. A ground pace vehicle with a

calibrated fifth-wheel was used as a speed reference. Surface wind conditions were 3 knots or less. Tests were flown IGE at a 10-foot skid height. The low-speed flight data are presented in figures 30 and 31, appendix E.

34. Control margins in low-speed forward and rearward flight were adequate. Lateral and directional control requirements for trimmed flight varied only 0.5 and 1.0 inches, respectively, throughout the test airspeed range from 30 KTAS rearward to 30 KTAS forward. There were no significant or unusual lateral or directional discontinuities throughout the airspeed range. The general trend of longitudinal control motion was satisfactory with forward movement of the cyclic required for increasing forward airspeed and aft cyclic movement required for increasing rearward speed. The minor nonlinearities and reversals shown in the test data were not objectionable to the pilot in flight. Within the scope of this test, the low-speed forward and rearward flight characteristics were unchanged from the Model 214A helicopter with the standard main rotor blades and are satisfactory.

35. During sideward flight, the variation of lateral control position with sideward airspeed was satisfactory with increasing right lateral control with increasing right sideward airspeed and vice versa, although unobjectionable minor reversals were present. Lateral control margins were adequate at all airspeeds. Minimum longitudinal control movement was required to achieve trimmed flight in either direction up to 10 KTAS. At left sideward airspeeds greater than 10 KTAS, a nose-down pitching moment was evidenced by the requirement for increasing aft longitudinal control as sideward velocity was further increased. Longitudinal trim shifts did not exist in right sideward flight. The trim shift in left sideward flight was not abrupt and did not cause a noticable pilot workload increase during the tests; however, this characteristic could pose difficulties during tasks requireing precision hover in left cross winds. This characteristics was observed during previous Model 214A tests (ref 10, app A). Within the scope of this test, the longitudinal and lateral control characteristic during sideward flight were unchanged from the Model 214A helicopter with the standard main rotor blades and are satisfactory.

36. During left sideward flight, the directional control margin was adequate. Directional control requirements with increasing sideward airspeed required increasing right pedal for increasing left sideward airspeed and vice versa, although unobjectionable minor reversals were present. The minimum directional control margin in left sideward flight was 2.5 inches (42 percent). In right sideward flight at the maximum airspeed tested, 15 KTAS, the directional control margin was 0.35 inches (6 percent). An increase in right sideward airspeed to 20 KTAS was not obtained because the left directional control limit was reached at approximately 18 KTAS. The inadequate directional control at airspeeds greater than 15 KTAS at high gross weight density altitude combinations is a deficiency which is not attributable to the fiberglass rotor blade installation. The gross weight and density altitude conditions for the left and right sideward flight tests were high (13,640 pounds and 10,120 feet, respectively). The Operator's Manual (ref 4, app A) does not contain any information on flight conditions where inadequate directional control exists. Consideration should be given to including flight conditions where inadequate directional control exists in the Operator's Manual similar to the manner in which it is presented in the Model 214B Helicopter Flight Manual (ref 12, app A). The directional control characteristics in sideward flight are unchanged from those previously reported.

AIRCRAFT SYSTEMS FAILURES

Simulated Engine Failures

37. Sudden engine failures were simulated in a variety of flight and power conditions by rapidly rolling the throttle to the idle stop. Following the simulated failure, all flight controls were held fixed until the main rotor speed decayed to 270 rpm on the cockpit indicator. The minimum transient rotor speed limit for this test was 249 rpm. Figures 32 through 34, appendix E are representative time histories of the simulated engine failures.

38. Aircraft response to simulated sudden engine failures in stabilized flight at all airspeed, gross weight, and power conditions tested was mild, repeatable, and predictable. Aircraft attitude perturbations were less than 10 degrees, and maximum angular rates were less than 10 degrees per second in any axis 2 seconds after the simulated engine failures at the higher airspeeds (116 and 140 KCAS). At 40 KCAS and power for level flight, the aircraft yawed 12 degrees before the collective was lowered (270 rpm) and at 46 KCAS and maximum continuous power (MCP), the maximum yaw rate was 16 degrees per second within 2 seconds after the simulated engine faulure. The rates and attitudes developed during simulated sudden engine failures were comfortable and could be easily controlled.

39. Collective control delay times greater than 1.5 seconds were not possible at all airspeeds tested at MCP. The rotor speed decreased below the operational minimum transient limit of 270 rpm in each of the 1.5-second delays at MCP, but the rpm readily increased to within the normal power-OFF operating range when the collective was lowered. The 2 second collective control delay time required by reference 6, appendix A was not met in every airspeed and power configuration tested; however, due to the ease with which the rotor rpm could be regained and good engine failure cues, the 1.5-second delay time was satisfactory.

40. The aural rpm warning tone which activated at 285 rpm was the primary cue of simulated sudden engine failure when initiated from power for level flight at all conditions tested. Aircraft reactions became more pronounced at high power and low airspeed conditions during which aircraft yaw was the initial cue of simulated sudden engine failure. Recovery into autorotational descent following simulated engine failures was easily accomplished (HQRS 2). Only minimal longitudinal, lateral, and directional control inputs were necessary to achieve satisfactory autorotational descent. The simulated engine failure characteristics of the Model 214A helicopter with the fiberglass rotor blades remain unchanged from those with the standard blades and are satisifactory.

STRUCTURAL DYNAMICS

Vibration

41. Vibration characteristics were quantitatively evaluted throughout the conduct of the evaluation. Representative level flight vibration data are presented in figures 35 through 42, appendix E.

42. In level flight, the Model 214A with the fiberglass main rotor provided a very smooth ride throughout the airspeed, altitude, and gross weight envelope tested. The 1/rev, 2/rev, and 4/rev vibration levels were slight (VRS 2) and always below the military specification limit of 0.15g single-amplitude acceleration and were generally below those observed on previous Model 214A helicopters (ref 10, app A). In one instance, the 6/rev vibration level exceeded the specification limit but was not objectionable. The vibration levels were below the levels previously recorded on Model 214A helicopters and are an enhancing characteristic. However, the lack of pilot vibration cues with higher airspeeds requires caution to not inadvertently exceed V_{NE}.

43. The vibration characteristics in maneuvering flight were similar to those reported on the Model 214A with standard rotor blades. Because the main rotor track and balance and nodal beam assembly tuning of the test aircraft were not optimized, the vibration levels increased rapidly with increasing load factor and were severe (VRS 9). After nodal beam tuning and main rotor tracking and balancing adjustments were made, the magnitude of the vibration levels was reduced but remained objectionable (VRS 7). The objectionable vibration levels during maneuvering flight at load factors beyond the capability of the nodal beam assembly remains a shortcoming.

44. BHT personnel spent considerable time tracking and balancing the main rotor to obtain and maintain a low level of 1/rev vibration. A track and balance procedure for the fiberglass rotor blade should be developed and evaluated.

CONCLUSIONS

GENERAL

45. Within the scope of this evaluation, the performance and handling qualities of the Model 214A helicopter with the fiberglass main rotor are similar to those of the standard blades. The following conclusions have been made:

a. The OGE hover gross weight capability is increased by 175 pounds to 13,975 pounds at 5000 feet pressure altitude and 35°C (para 9).

b. The 10-foot hover gross weight capability is increased by 459 pounds to a gross weight of 15,532 pounds at 5000 feet pressure altitude and 35°C (para 9).

c. Maximum level flight true airspeed is 155 KTAS at 5000 feet pressure altitude, standard day conditions, 13,000 pound gross weight and maximum continuous power, which exceeds the 144-knot guarantee and is above the never-exceed velocity, V_{NE} (para 11).

d. Autorotational performance is unchanged (para 13).

e. One enhancing characteristic and one shortcoming which were identified previously, are restated because of their significance.

f. One deficiency not attributable to the fiberglass rotor installation and not previously identified was noted.

Enhancing Characteristic

46. The low level-flight vibration levels (para 42).

Deficiency

47. The inadequate directional control at airspeeds greater than 15 KTAS in right sideward flight at high gross weight, density altitude combinations (para 36).

RECOMMENDATIONS

49. The enhancing characteristic should be incorporated in future designs (para 6).

50. The deficiency must be corrected (para 47).

51. The shortcoming should be corrected (para 48).

52. Include flight conditions in the Operator's Manual where inadequate directional control exists (para 36).

53. Develop and evaluate a track and balance procedure for the fiberglass rotor blade (para 44).

APPENDIX A. REFERENCES

1. Letter, AVRADCOM, DRDAV-EQI, 26 September 1977, subject: Limited Airworthiness and Flight Characteristics of the 214 A/C Helicopter Incorporating a Fiberglass Main Rotor Blade.

2. Test Plan, USAAEFA Project No. 77-32, *Limited Airworthiness and Flight Characteristics Evaluation, 214A/C Helicopter Incorporating a Fiberglass Main Rotor Blade*, February 1978.

3. Letter, AVRADCOM, DRDAV-EQ, 11 May 1978, subject: Advanced Test Plan: Limited Airworthiness and Flight Characteristics Evaluation 214A/C Helicopter Incorporating a Fiberglass Main Rotor Blade.

4. Interim Technical Manual, ITM 55-1520-232-10, *Operator's Manual, Imperial Iranian Army Model 214A Helicopter*, 21 April 1978.

5. Letter, AVRADCOM, DRDAV-EQ, 25 August 1978, subject: Airworthiness Release for USAAVRADCOM/USAAEFA Project No. 77-32.

6. Military Specification, MIL-H-8501A, *Helicopter Flying and Ground Handling Qualities; General Requirements For*, 7 September 1961, with Amendment 1, 3 April 1962.

7. Engineering Design Handbook, Army Material Command, AMCP 706-204, *Helicopter Performance Testing*, August 1974.

8. Flight Test Manual, Naval Air Test Center, FTM No. 102, *Helicopter Performance Testing*, 28 June 1968.

9. Flight Test Manual, Naval Air Test Center FTM No. 101, *Helicopter Stability and Control*, 10 June 1968.

10. Final Report, United States Army Aviation Engineering Flight Activity, USAAEFA Project No. 74-30, *Airworthiness and Flight Characteristics Evaluation, Iranian Model 214A Helicopter*, June 1975.

11. Final Report, United States Army Aviation Engineering Flight Activity, USAAEFA Project No. 76-02, *Performance and Systems Evaluation, Iranian Model 214C Helicopter*, March 1977.

12. Flight Manual, Bell Helicopter Textron, *Bell Model 214B*, 27 January 1976.

13. Executive Summary, Bell Helicopter Textron, *Fiberglass Main Rotor Blade, Model 214*, 15 April 1977.

APPENDIX B. AIRCRAFT DESCRIPTION

GENERAL

1. The Model 214A helicopter (photos 1 and 2) is a utility helicopter built by Bell Helicopter Textron and is a derivative of the UH-1 series aircraft. The aircraft used in this evaluation, SN 27001 (N 214 J), was essentially a Model 214A. The primary differences between the test aircraft and a Model 214A helicopter were the installation of a smaller capacity fuel system, a rotor brake, an engine compartment fire extinguisher system, a engine high exhaust gas temperature (EGT) monitoring device, an engine overspeed trip system, a nose-mounted pitot system, a single direct current (dc) generator and a different external antenna configuration SCAS actuator authority was reduced from that of a Model 214A (ref 10, app A) to ±13 percent in pitch, ±10 percent in roll, and ±10 percent in yaw. A complete description of the Model 214A helicopter is contained in reference 4, appendix A. Major features of the helicopter are described below.

Main Rotor

2. The main rotor system is a two-bladed, semi-rigid, teetering type employing preconing and underslinging. A typical planform and cross section of the fiberglass main rotor blades is shown in figure 1 and photo 3, respectively. The spar caps, trailing edge strip, spar tubes and overwrap are constructed with fiberglass. The inboard half of the noseblock is made of syntactic foam and fiberglass. The weight and inertia characteristics of the fiberglass blades are similar to the metal blades. A lead noseblock is used to provide rotational inertia in the outboard half of the blade. The leading edge of the noseblock is covered by a titanium abrasion strip. The blade afterbody consists of a nomex honeycomb core wrapped with fiberglass skins. The blades incorporate a 33-inch chord Wortmann FX 69-H-098 airfoil with 70-degree swept tips and an 8-degree negative twist, which are the same as the metal blades. Each blade is connected to a common yoke by grip pitch change bearings with tension-torsion straps to carry centrifugal loads. The yoke utilizes elastomeric bearings for flapping and oil-lubricated steel needle bearings for feathering. The blades are interchangeable and incorporate a trim tab on the trailing edge. Main rotor dimensions are shown in figure 1. A complete description of the rotor blade is contained in the BHT Executive Summary, Fiberglass Main Rotor Blade (ref 13, app A).

Rotor Brake

3. A hydro-mechanical rotor brake system is incorporated on the main transmission. The master cylinder and actuation lever are mounted above and to the left of the pilot's station. The rotor brake is used during shutdown to reduce rotor coast-down time.

Engine

4. The Lycoming T5508D engine installed in the test aircraft is a direct-shaft turbine incorporating a combination axial/centrifugal compressor and an annular-type combustor. It has the following thermodynamic rating: (1) Takeoff (10 min-

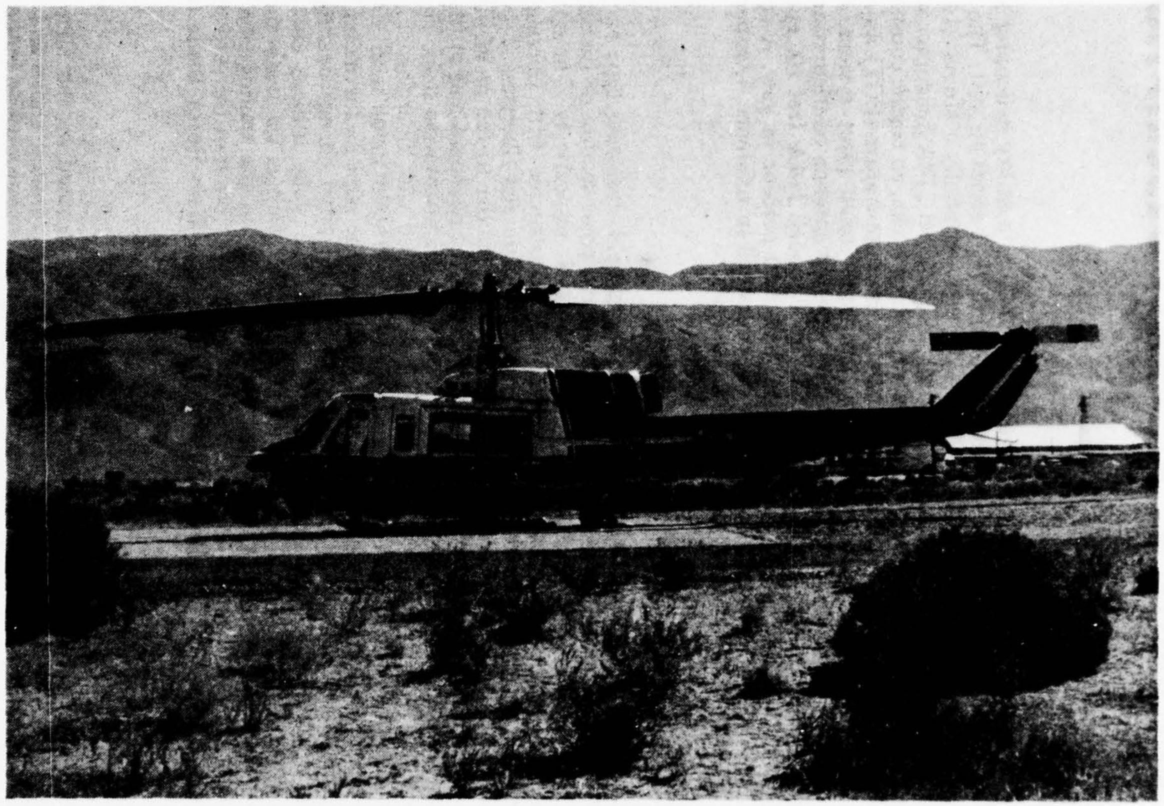


Photo 1. Model 214A Helicopter (left side).

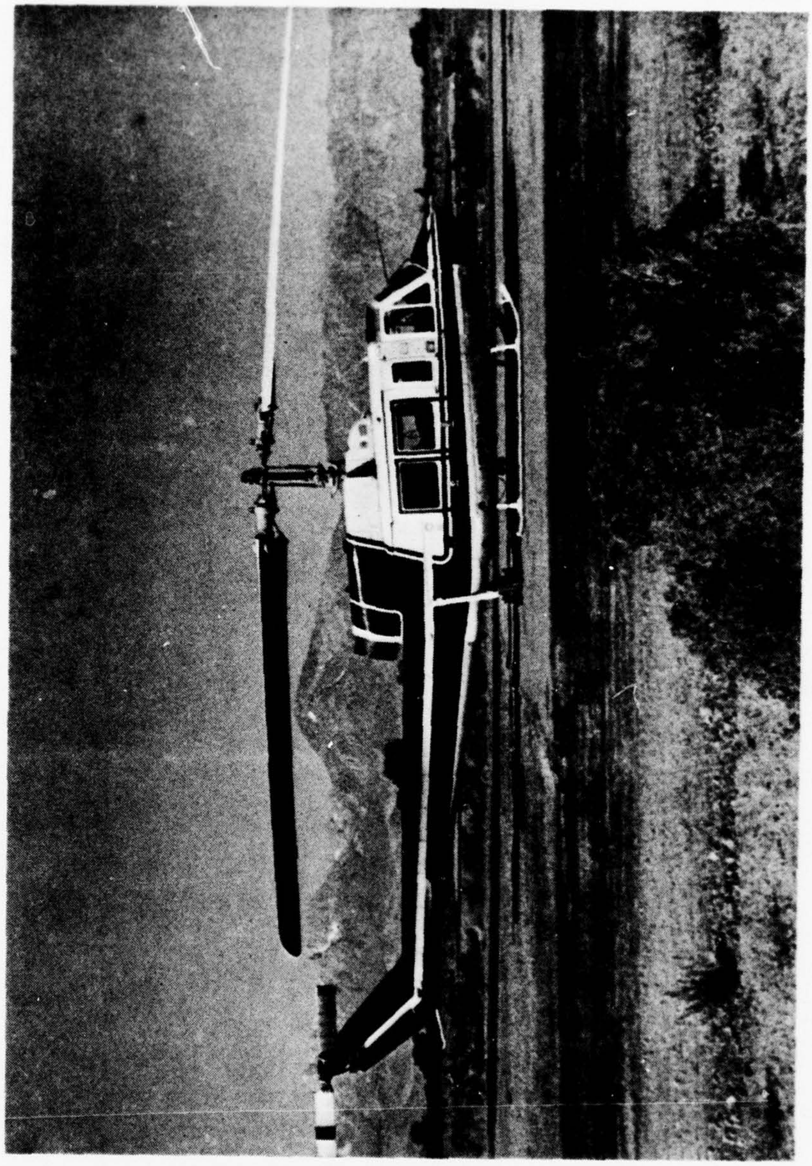


Photo 2. Model 214A Helicopter (right side).

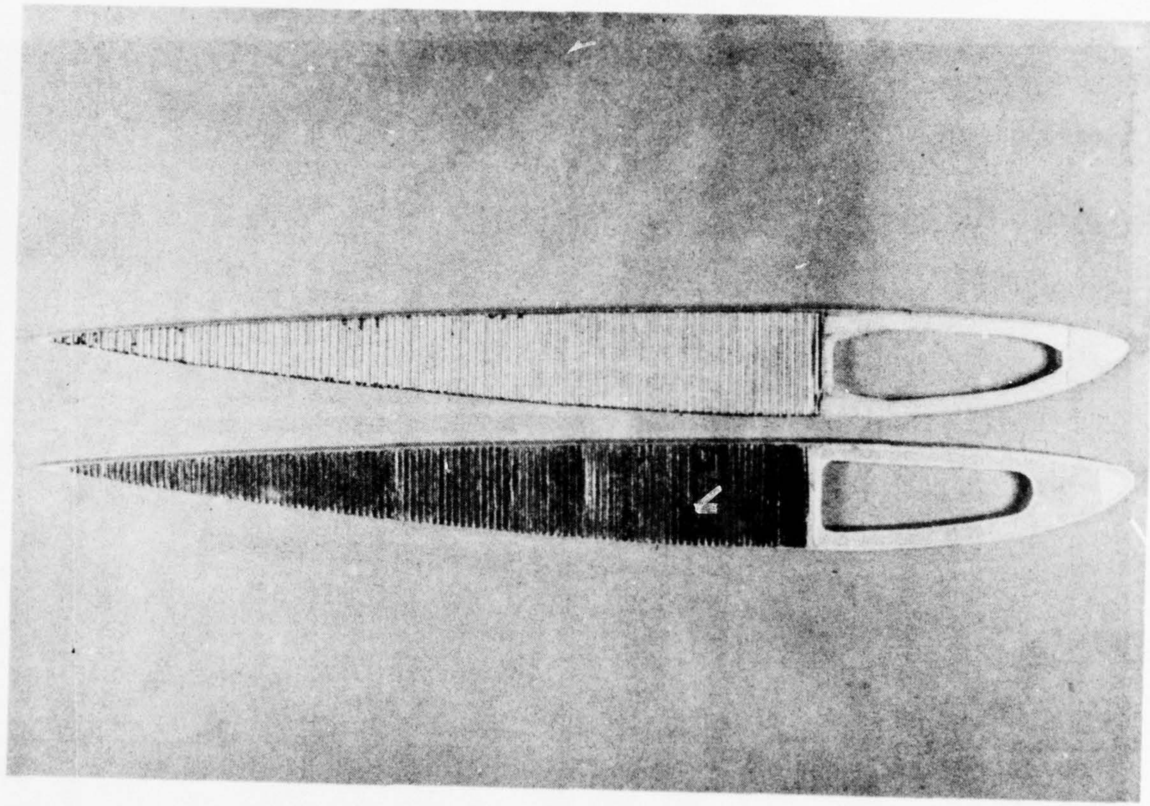


Photo 3. Model 214A Helicopter main rotor blade cross section (upper - metal, lower - fiberglass).

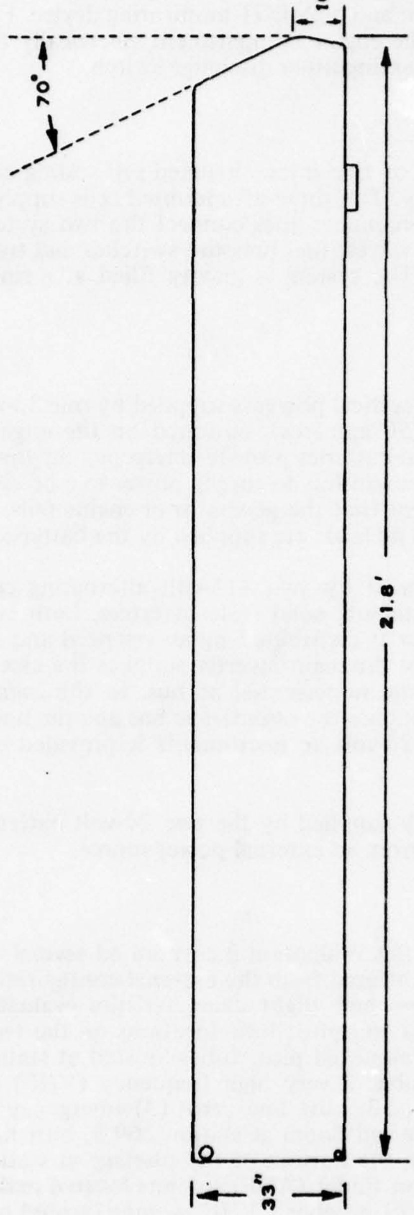


Figure 1. 214 Fiberglass Main Rotor Blade Planform.

utes), 2930 shp; (2) Intermediate (30 minutes), 2735 shp; (3) Maximum Continuous, 2500 shp, but is derated to 2250 shp for the Model 214A installation. The engine incorporates an electrically-operated anti-icing system, a particle separator, an engine overspeed trip system and high EGT monitoring device. Fire extinguishing agent can be discharged into the engine compartment electrically from the cockpit by actuation of the engine fire extinguisher discharge switch.

Fuel System

5. The fuel system consists of five interconnected self-sealing crash-worthy fuel cells with a 210 gallon capacity. The three aft-mounted cells supply fuel to the two forward lower main cells. Interconnect lines connect the two systems. The system includes a shutoff valve, drain valves, fuel pressure switches and transmitters, quantity gage, and caution lights. The system is gravity filled at a single point on the right side of the helicopter.

Electrical System

6. The primary 28-volt dc electrical power is supplied by one 30-volt, 300 ampere starter-generator (derated to 250 apmeres), mounted on the engine. Two 24-volt, 40 ampere-hour nickel cadmium batteries provide emergency dc power. The primary dc power is distributed by the main bus to supply power to a dc essential bus and a dc nonessential bus. In the event that the generator or engine fails, the nonessential bus is dropped, and all essential dc loads are supplied by the batteries.

7. Secondary power is supplied by two 115-volt alternating current (ac), 400 hertz, single-phase, 250 volt-ampere, solid state inverters, both connected to the essential dc bus. This ac power is distributed by an essential and nonessential bus arrangement such that normally the main inverter supplies the essential ac bus and the standby inverter supplies the nonessential ac bus. In the event of an inverter failure, the operable inverter supplies the essential ac bus and the nonessential ac bus is deactivated. Power for the 26-volt ac instruments is provided by a transformer supplied by the essential bus.

8. Starting power is normally supplied by the two 24-volt batteries. The aircraft also has provisions for starting from an external power source.

External Configuration

9. The Model 214A used in this evaluation incorporated several external antenna and pitot tube changes which differed from the external configuration of the Model 214A used in the airworthiness and flight characteristics evaluation, USAAEFA Project No. 74-30. The antenna and pitot tube locations on the test aircraft are as follows (fig. 2): (1) dual nose mounted pitot tubes located at station 26, butt line 17.75, left and right; (2) number 2 very high frequency (VHF) antenna located beneath the fuselage at station 53, butt line zero; (3) emergency locator antenna located on the right side of the tail boom at station 269.5, butt line 17, right; (4) transponder antenna located on the bottom of the fuselage at station 21, butt line 1.5, left; (5) automatic direction finder (ADF) antenna located under the tail boom at station 239, butt line zero; (6) number 1 VHF antenna located on the cabin roof at station 65, butt line zero; (7) VHF omnidirectional range (VOR) antenna located on the tail boom at station 437; (8) ADF loop antenna located beneath the fuselage at station 190, butt line zero.

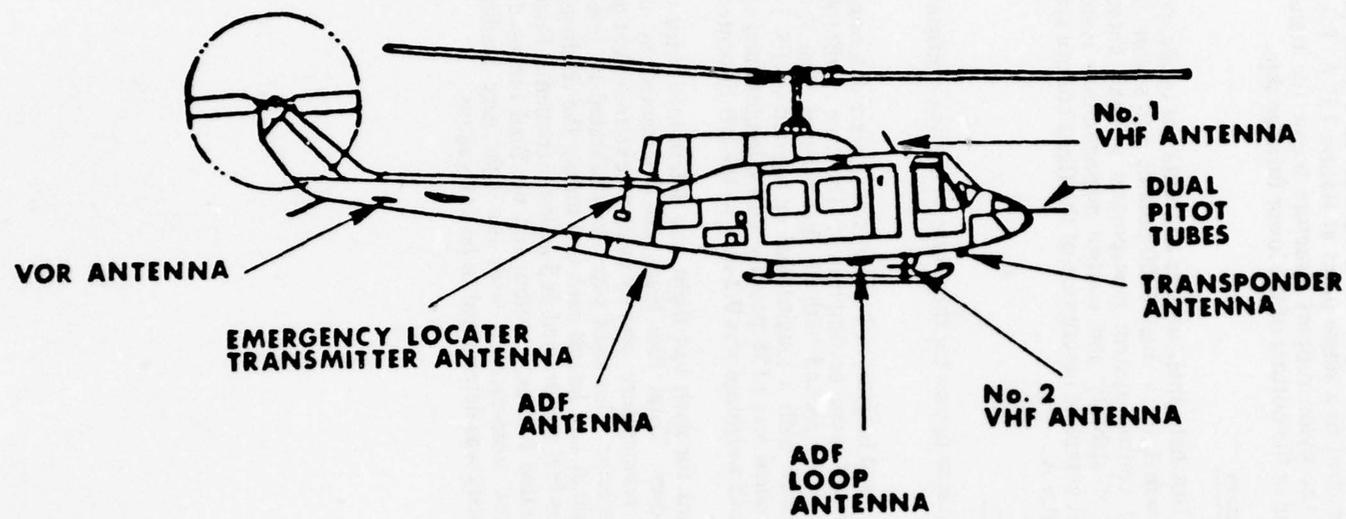


Figure 2. External Configuration.

10. The cargo suspension system is comprised of a 6000 pound capacity cargo hook assembly attached to a single point at station 137.6. The hook is attached to the lift beam in the pylon support structure below the transmisssion, extending through an opening in the bottom of the lower fuselage skin.

Flight Control System

11. The Model 214A helicopter utilizes conventional cyclic, collective, and directional controls powered by a dual 3000-pounds per square inch (psi) hydraulic system. The flight control system incorporates a 3-axis automatic flight control system consisting of stability and control augmentation system and an attitude retention system. A complete description of the flight control system is contained in reference 4, appendix A.

Weight and Balance

12. Weight and balance figures for three typical loading configurations are shown in table 1.

13. Aircraft weight and longitudinal and lateral cg were determined prior to testing. Two aircraft weighings were accomplished: The first with full fuel and aircraft instrumentation, and the second with all fuel drained. The first weight with full fuel was 9547 pounds with a longitudinal cg located at FS 149.87. The second weight with fuel drained was 8198 pounds with a longitudinal cg of 149.32 inches. The lateral cg for both weighings was 0.2-inch to the left of center line.

14. The fuel aboard for each test flight was determined prior to engine start and after engine shutdown. Total fuel load was determined by measuring the fuel specific gravity and temperature, and by using an external sight gage on the fuel cell to determine fuel volume. This sight gage was calibrated by leveling the helicopter through its longitudinal and lateral axes and noting the readings on the externally attached sight gage as fuel was drained in 5-gallon increments. Fuel used in flight and recorded by a calibrated fuel-used system and the final fuel-used reading following engine shutdown was cross-checked with the sight gage readings following each flight. The fuel capacity was determined to be 210 gallons.

Table 1. Weight and Balance.

Configuration	Weight (lb.)	Center-of-Gravity Location[1] (lb.)
Basic aircraft, 2 pilots[2], full fuel (JP-4)	10,026	145.02
Basic aircraft, 2 pilots, 14 passengers[3], 274 pounds cargo, full fuel (JP-4)	13,800	137.56
Basic aircraft, 2 pilots 4974 pound load on cargo hook full fuel (JP-4)	15,000	140.9

[1] Longitudinal center-of-gravity limits: 132.5 to 147 inches; lateral center-of-gravity limits: 4.7 inches left to 4.7 inches right.
[2] Pilot/copilot weight is 235 pounds per detail specification.
[3] Passenger weight is based on 240 pounds per passenger.

APPENDIX C. INSTRUMENTATION

1. The test instrumentation was installed, calibrated, and maintained by BHT. A test airspeed boom with swiveling pitot-static head connected to an airspeed indicator and altimeter were installed at the nose of the aircraft. Data from the following aircraft sources were obtained from calibrated instrumentation and displayed or recorded as indicated below.

Pilot Panel

Airspeed (boom)
Altitude (boom)
Altitude (radar)*
Rate of climb
Rotor speed
Engine torque*
Exhaust gas temperature*
Gas generator speed (N_1)*
Power turbine speed (N_2)*
Control position:
    Longitudinal
    Lateral
    Directional
    Collective
Center-of-gravity normal acceleration
Angle of sideslip
Angle of attack
Pitch attitude*
Roll attitude*
Time of day*
Turn-and-slip indicator*
Load position (cable angle) indicator

Engineer Panel

Record counter
Magnetic tape control panel
Event marker
Airspeed*
Altitude*
Rotor speed*
Outside air temperature
Fuel flow rate
Fuel-used totalizer
Cable tension
Engine torque*

*Standard ship's instrument

Magnetic Tape

Airspeed (boom)
Airspeed*
Altitude (boom)
Altitude (radar)
Outside air temperature
Rotor speed
Gas generator speed (N_1)
Power turbine speed (N_2)
Engine torque
Exhaust gas temperature
Main rotor shaft torque
Tail rotor shaft torque
Fuel flow
Fuel-used totalizer
Fuel temperature
Event marker
Control positions:
- Longitudinal
- Lateral
- Directional
- Collective
- Throttle

Longitudinal control force
SCAS positions:
- Longitudinal
- Lateral
- Directional

Attitude:
- Pitch
- Roll
- Yaw

Rate:
- Pitch
- Roll
- Yaw

Acceleration:*
- Vertical (pilot's seat, copilot's seat, aircraft cg)
- Lateral (pilot's seat, copilot's seat, aircraft cg)
- Fore/aft (pilot's seat, copilot's seat, aircraft cg)

Main rotor flapping
Tail rotor blade angle
Angle of attack
Angle of sideslip
Voice
Cable tension
Time code
Main rotor azimuth index
Tail rotor azimuth index

*Standard ship's instrument.

BASIC AIRCRAFT INFORMATION

15. Additional aircraft descriptive data are shown in the following listing.

Main Rotor

Diameter	50 ft
Chord	33 in.
Disc area	1963.5 ft^2
Number of blades	2
Solidity ratio	0.070
Disc loading (13,000-lb gross wt)	6.62 lb/ft^2
Blade twist	-8 deg
Airfoil section	Wortmann FX69-H-098
Flapping angle	±10 deg
Hub precone angle	±2.5 deg
Design rotor speed (100 percent)	300 rpm
Design tip speed	785 ft/sec

Tail Rotor

Diameter	9 ft, 8 in.
Chord	12 in.
Number of blades	2
Solidity ratio	0.132
Blade twist	zero deg
Airfoil	Wortmann FX69-H-083/10 MOD
Hub precone angle	1.0 deg
Design rotor speed (100 percent)	1601 rpm
Design tip speed	810 ft/sec

Vertical Tail

Span	54 in.
Chord, average	44.5 in.
Area	15.6 ft^2
Leading edge sweep angle	46 deg
Airfoil section	BHC cambered per 214-E106 line drawing

Horizontal Tail

Span	113.28 in.
Chord	30.6 in.
Area	19.8 ft^2
Leading edge sweep angle	zero deg
Airfoil section	Inverted Clark Y

Landing Gear

Width (no load)	100 in. (between skid center lines

Weight

Basic aircraft	7831 lb
Maximum gross weight, internal load	13,800 lb
Maximum gross weight, external load	15,000 lb

Engine (one)

Takeoff rating (thermodynamic)	2930 shp
Maximum Continous rating (thermodynamic)	2500 shp
Rated output shaft speed	14,750 rpm

Speed Reduction Gearbox

Engine input	14,695 rpm
Reduction gearbox output	7237 rpm

Main Transmission

Shaft horsepower rating (5 minutes)	2050 shp
Shaft horsepower (continuous)	1845 shp
Design input shaft speed	7237 rpm
Design mast speed	300 rpm

Flight Airspeeds

V_{NE} at sea level (not power limted)	140 KCAS
Sideward flight	35 KCAS
Rear ward flight	30 KCAS
V_{NE} sling load	100 KCAS

Main Rotor Speeds

Normal operating	100 percent (300 rpm)
Power-on continuous	98 to 100 percent (294 to 300 rpm)
Power-on transient	105 percent (315 rpm)
Power-off continuous	95 to 105 percent (285 to 315 rpm)
Power-off transient	90 to 105 percent (270 to 315 rpm)

APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

TEST TECHNIQUES

Level Flight Performance

1. Level flight performance parameters were determined utilizing the constant ratio of weight-to-density ratio (W/σ) method described in AMCP 706-204 (ref 7, app A). This method allows data to be gathered at a constant value of the non-dimensional parameter C_T, defined in paragraph. The constant weight to pressure ratio (w/δ) constant referred rotor speed ($N/\sqrt{\theta}$) method was planned for this test (ref, app A), but was not possible because of the limited power-ON rotor speed range of the Model 214A helicopter. The aircraft was stabilized at zero sideslip at airspeeds between 40 KCAS and V_H or V_{NE}. All test points were flown for a minimum of 2 minutes at each stabilized test condition.

Autorotational Descent Performance

2. Autorotational descent performance tests were conducted by stabilizing at engine flight idle with needles spilt in an autorotational descent at constant airspeed and rotor speed at zero degree sideslip. Rotor speed was maintained at 300 rpm by adjusting collective position. Rate of descent was determined by recording time to descent through 1000 feet. The test was repeated at various airspeeds. Rotor speed was varied at minimum rate of descent airspeed to determine the effect of rotor speed on rate of descent.

Control Positions in Trimmed Forward Flight

3. Control positions in trimmed forward flight were evaluated in conjunction with level flight performance tests. Data were obtained by stabilizing at zero sideslip at 10-knot increments, trimming control forces to zero, and recording control position.

Collective-Fixed Static Longitudinal Stability

4. Data were obtained by trimming the aircraft in ball-centered level flight at the desired airspeed and securing the collective control in that position. Airspeed was then varied ±20 knots from trim in 5-knot increments, utilizing the cyclic and directional controls only, and allowing altitude to vary as necessary. Control positions were recorded at each airspeed.

Static Lateral-Directional Stability

5. Tests were conducted in level flight by trimming the aircraft at the desired airspeed and securing the collective control. Data were obtained by varying sideslip angle incrementally to the limits of the sideslip envelope. Collective position, airspeed, and aircraft ground track were held constant and altitude allowed to vary as required. Control positions and aircraft attitude were recorded at each stabilized point.

Maneuvering Stability

6. Maneuvering stability tests were accomplished by initially stabilizing the helicopter in zero sideslip level flight at the desired airspeed and recording the trim condition. Load factor was then increased by stabilizing the helicopter at increasing bank angles in left and right turns. Airspeed, collective position and coordinated flight were maintained and altitude allowed to vary throughout all the maneuvers.

Dynamic Stability

7. Tests were initiated in ball-centered level flight and a hover. Data were obtained by evaluating the aircraft motions that resulted from pulse-type inputs about the longitudinal, lateral, and directional axes. Each input was accomplished by rapidly displacing the particular control approximately 1 inch from trim, holding in this position for 0.5-second, then rapidly returning to the trim position and holding control fixed until aircraft motions were damped or corrective action became necessary. All controls other than the input control remained fixed during the test. Long-term dynamic response in forward flight was observed by displacing the aircraft from the trim airspeed using longitudinal cyclic. When an airspeed change of 10 ~ 15 KIAS was achieved, the control was returned to the trim position and held fixed at trim while the response of the aircraft was observed. Tests were conducted both SCAS ON and OFF.

Controllability

8. Tests were initiated in ball-centered level flight and a hover. Single-axis control step inputs of several magnitudes from ¼-inch to 1 inch were applied to the longitudinal, lateral, and directional controls, using mechanical fixtures to obtain the desired control input size. The step inputs were held steady while other controls were maintained at the trim control position and the subsequent aircraft angular displacement, after one second, maximum angular rate, and maximum angular acceleration were recorded. Tests were conducted both SCAS ON and OFF.

Simulated Engine Failures

9. Tests were initiated in ball-centered level and climbing flight by rapidly closing the throttle to the flight-idle position to simulate an engine failure. Following the simulated engine failure, all flight controls were held fixed until collective application was necessary to maintain rotor speed within established limits. Aircraft response subsequent to a sudden engine failure and the capability of the aircraft to transition safely into power-off autorotation were evaluated.

Vibration Characteristics

10. Aircraft vibrations were evaluated throughout the test program. Noise, visual cues, and touch were used as qualitative criteria to aid in evaluation and determining vibration amplitude. A fast Fourier transform harmonic analysis was conducted on the vibration data to determine the 1, 2, 4 and 6 per revolution harmonic content. A Vibration Rating Scale (VRS) (fig. 2) was used to augment pilot comments relative to vibration.

Pitot-Static System Calibration

11. Calibration of the boom airspeed system was accomplished in level, climbing, and descending flight with a trailing bomb. The aircraft was stabilized at the desired condition, and data was simultaneously recorded from the test aircraft and the trailing bomb. In all flight regimes, data were obtained by stabilizing in ball-centered flight in 10-knot increments throughout the desired airspeed range.

Data Analysis Methods

12. The helicopter performance test data were generalized by use of nondimensional coefficients. The following nondimensional coefficients were used to generalize the level flight results obtained during this flight test program.

a. Coefficient of power (C_P):

$$Cp = \frac{SHP\ (550)}{\rho A(\Omega R)^3} \quad (1)$$

b. Coefficient of thrust (C_T):

$$C_T = \frac{W}{\rho A(\Omega R)^2} \quad (2)$$

c. Advance ratio (μ):

$$\mu = \frac{1.6876\ V_T}{\Omega R} \quad (3)$$

d. Advancing tip Mach number (M_{tip}):

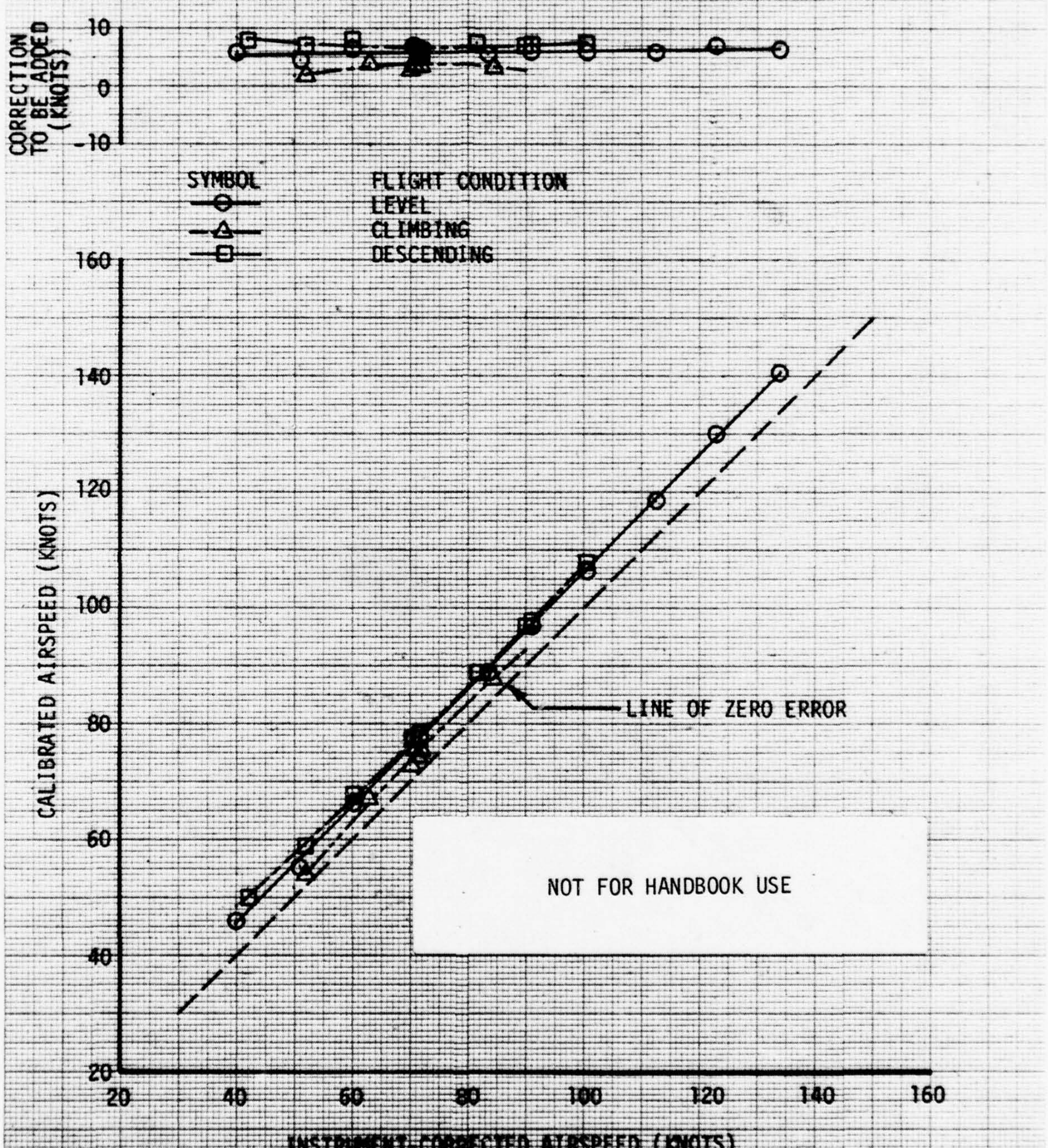
$$M_{tip} = \frac{1.6878\ V_T + (\Omega R)}{a} \quad (4)$$

Where:

SHP = Output shaft horsepower.

550 = Conversion factor (ft-lb/sec/shp).

FIGURE 1
AIRSPEED CALIBRATION
BHT MODEL 214
S/N 27001
BOOM SYSTEM



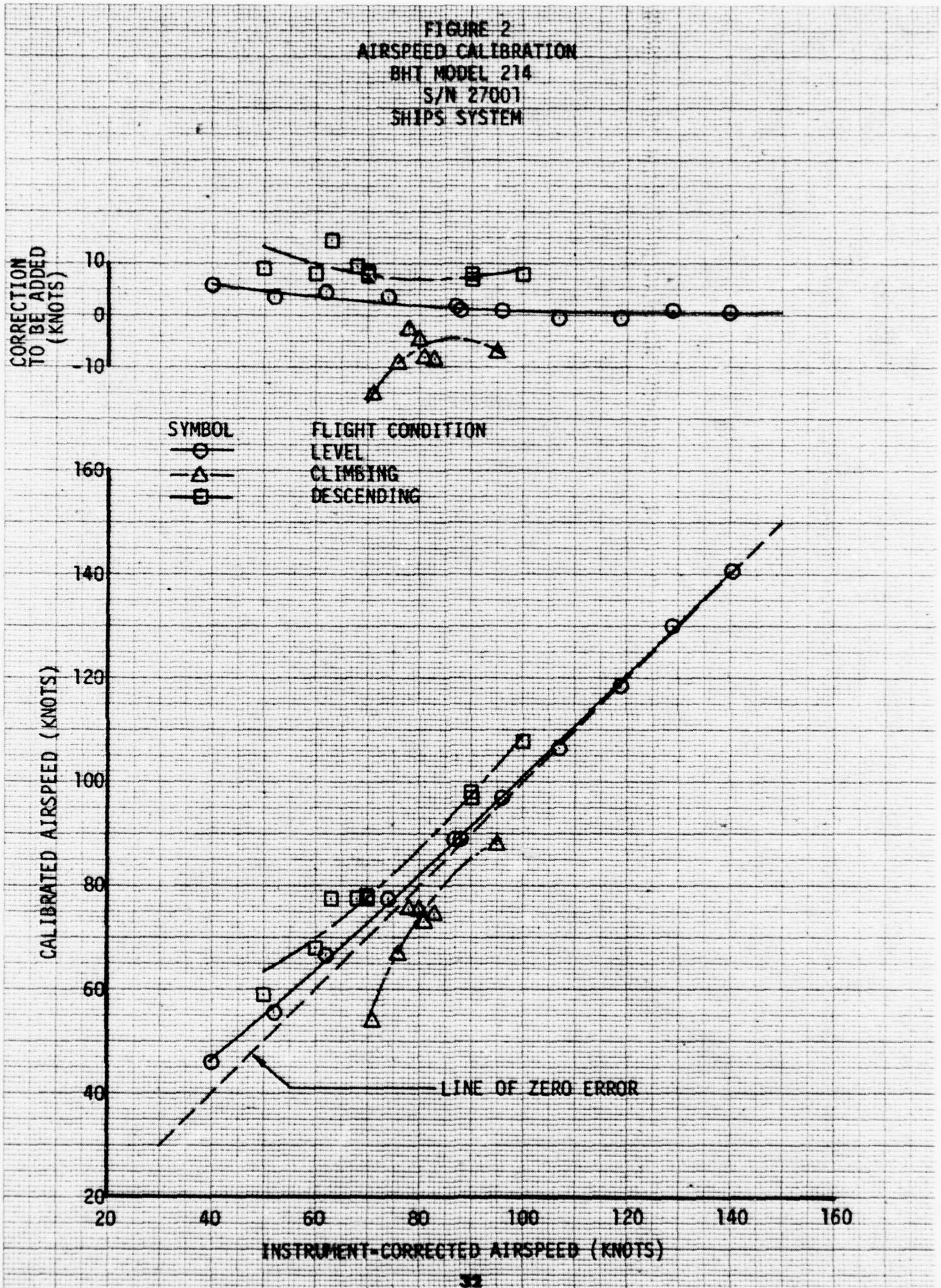


FIGURE 2
AIRSPEED CALIBRATION
BHT MODEL 214
S/N 27001
SHIPS SYSTEM
CORRECTION TO BE ADDED (KNOTS)
10
0
-10
SYMBOL
FLIGHT CONDITION
LEVEL
CLIMBING
DESCENDING
CALIBRATED AIRSPEED (KNOTS)
160
140
120
100
80
60
40
20
LINE OF ZERO ERROR
20
40
60
80
100
120
140
160
INSTRUMENT-CORRECTED AIRSPEED (KNOTS)

ρ = Air density (slug/ft^3) = $2.3769 \times 10^{-3}\ \sigma$.

A = Main rotor disc area (ft^2) = πR^2.

Ω = Main rotor angular velocity (rad/sec) = $\frac{\pi}{30}$ = x RPM.

R = Main rotor radius (ft).

W = Aircraft gross weight (lb).

1.6878 = Conversion factor (ft/sec/kt).

V_T = True airspeed (kt).

a = Speed of sound (ft/sec) = $1116.45\sqrt{\theta}$.

σ = Air density ratio = δ/θ.

$$\delta = \text{Pressure ratio} = \left(1 - \frac{Hp}{145{,}422}\right)^{5.25585} \tag{5}$$

Hp = Pressure altitude (ft).

$$\theta = \text{Temperature ratio} = \frac{OAT + 273.15}{288.15} \tag{6}$$

$$OAT = \text{Ambient air temperature } (^\circ C) = \frac{(OAT_{ic} + 273.15)}{(1 + .2M)^2} \tag{7}$$

OAT_{ic} = Observed free air temperature corrected for instrument error (°C).

$$M = \text{Mach number} = \frac{V_{cal}}{661.49\sqrt{\delta}} \tag{8}$$

V_{cal} = Calibrated airspeed (kt).

For a rotor speed of 300 rpm, the following constants were used:

R = 25 ft.

A = 1.963495408×10^3 ft^2.

$\Omega R = 7.853981634 \times 10^2$ ft/sec.

$(\Omega R)^2 = 6.168502750 \times 10^5$ ft^2/sec^2.

$(\Omega R)^3 = 4.844730731 \times 10^8$ ft^3/sec^3.

13. The engine output shaft torque was determined from the engine manufacturer's torquemeter system which was displayed on cockpit indicators in units of percent of maximum transmission torque. The relationship of indicator reading (percent of torque) and input signal (VDC) was determined from a calibration of the indicator accomplished by BHT. The relationship of torquemeter output (*ie*., indicator input signal to actual engine output shaft torque (ft-lb) was determined from a Lycoming calibration of the torque transducer during the engine green run. Tabular data from the engine and indicator calibrations is presented in figure 3. The output shp was determined from the engine output shaft torque and rotational speed by the following equation:

$$SHP = \frac{N_P \frac{\pi}{30} \times Q}{550} = \frac{N_P \times Q}{5252} \qquad (9)$$

Where:

N_P = Engine output shaft rotational speed (rpm).

Q = Engine output shaft torque (ft-lb).

5252 = Conversion factor (rpm x ft-lb/min/shp).

14. The measured power required for level flight was corrected to standard conditions by assuming that the nondimensional power parameter, C_P, is a function of C_T only, and is independent of the gross weight and density altitude used to obtain the C_T. From equation 1, the following relationship can be derived:

$$SHP_s = SHP_t \times \frac{\rho_s}{\rho_t} \qquad (10)$$

Where:

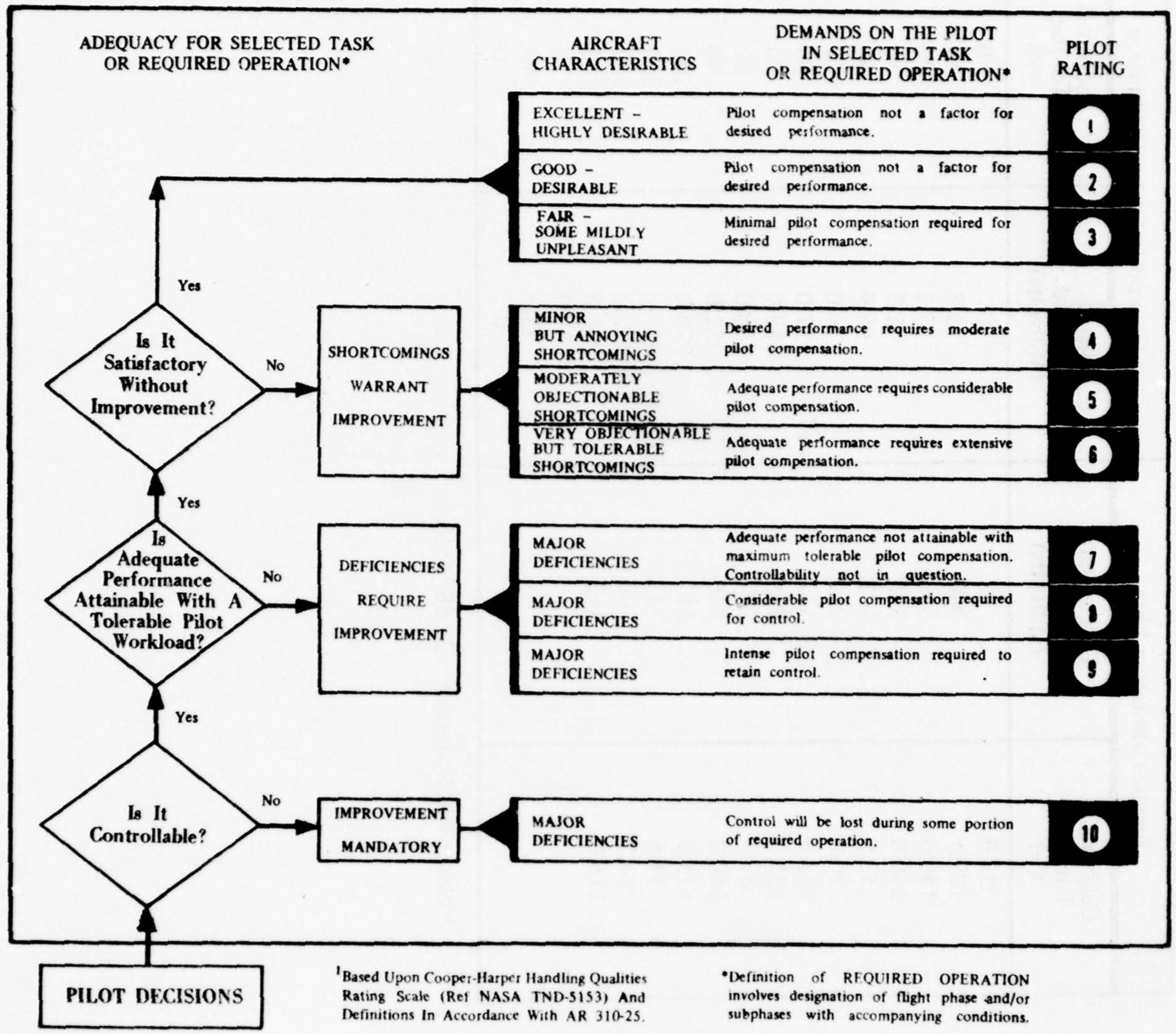
t = Test day.

s = Standard day.

Figure 3. Engine Torquemeter System Calibration

Engine Calibration Data		Cockpit Indicator Calibration	
Calibrated Torque* (%)	Torquemeter Output (Milivolts)	Signal Applied (Milivolts)	Indicated Torque (%)
37.9	98	0	0
109.5	300	21.8	10
140.0	343	43.2	20
140.0	343	72.3	30
53.5	143	95.9	40
90.5	254	126.0	50
123.8	322	157.0	60
135.9	338	189.0	70
67.0	187	220.0	80
140.2	345	252.0	90
115.2	311.7	278.0	100
11.6	32.3	303.0	110
34.7	84.9	322.0	120
116.0	312	278.0	100
		189.0	70
		95.9	40
		0	0

* Note: 100% torque = 732.687 ft-lb.



[1]Based Upon Cooper-Harper Handling Qualities Rating Scale (Ref NASA TND-5153) And Definitions In Accordance With AR 310-25.

*Definition of REQUIRED OPERATION involves designation of flight phase and/or subphases with accompanying conditions.

Handling Qualities Rating Scale

DEGREE OF VIBRATION	DESCRIPTION[1]	PILOT RATING
No vibration		0
Slight	Not apparent to experienced aircrew fully occupied by their tasks, but noticeable if their attention is directed to it or if not otherwise occupied.	1 2 3
Moderate	Experienced aircrew are aware of the vibration but it does not affect their work, at least over a short period.	4 5 6
Severe	Vibration is immediately apparent to experienced aircrew even when fully occupied. Performance of primary task is affected or tasks can only be done with difficulty.	7 8 9
Intolerable	Sole preoccupation of aircrew is to reduce vibration level.	10

[1] Based upon the Subjective Vibration Assessment Scale developed by the Aeroplane and Armament Experimental Establishment, Boscombe Down, England.

Vibration Rating Scale

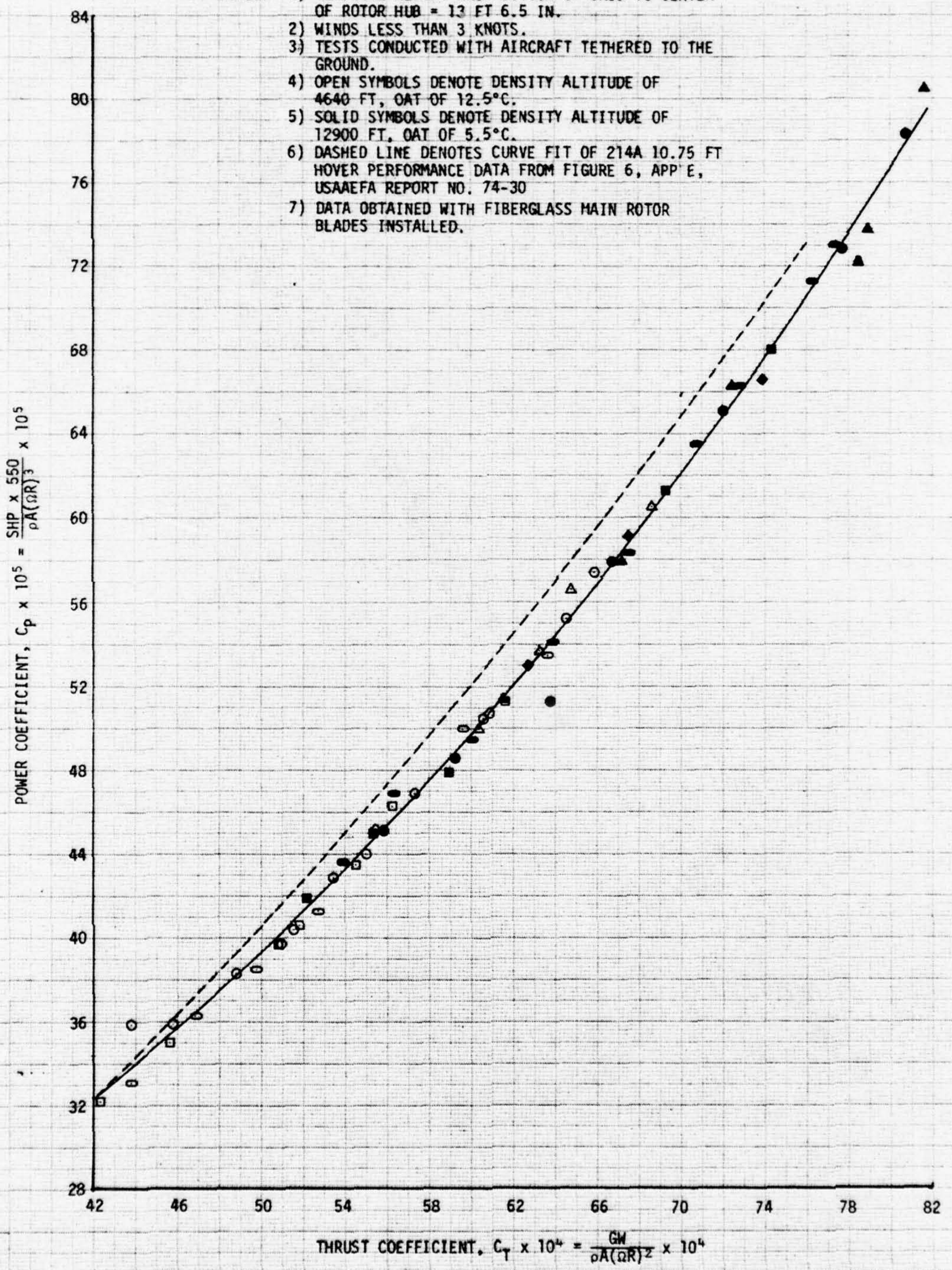
APPENDIX E. TEST DATA

INDEX

FIGURE 1
NONDIMENSIONAL HOVER PERFORMANCE
BELL MODEL 214 S/N 27001
LYCOMING ENGINE MODEL T5508D S/N D-8
SKID HEIGHT = 10 FT 1 INCH

SYMBOL	ROTOR SPEED (RPM)
△	281-283
△	284-286
○	288-290
▭	293-295
◇	296-298
□	299-302

NOTES:
1) VERTICAL HEIGHT FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB = 13 FT 6.5 IN.
2) WINDS LESS THAN 3 KNOTS.
3) TESTS CONDUCTED WITH AIRCRAFT TETHERED TO THE GROUND.
4) OPEN SYMBOLS DENOTE DENSITY ALTITUDE OF 4640 FT, OAT OF 12.5°C.
5) SOLID SYMBOLS DENOTE DENSITY ALTITUDE OF 12900 FT, OAT OF 5.5°C.
6) DASHED LINE DENOTES CURVE FIT OF 214A 10.75 FT HOVER PERFORMANCE DATA FROM FIGURE 6, APP E, USAAEFA REPORT NO. 74-30
7) DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.



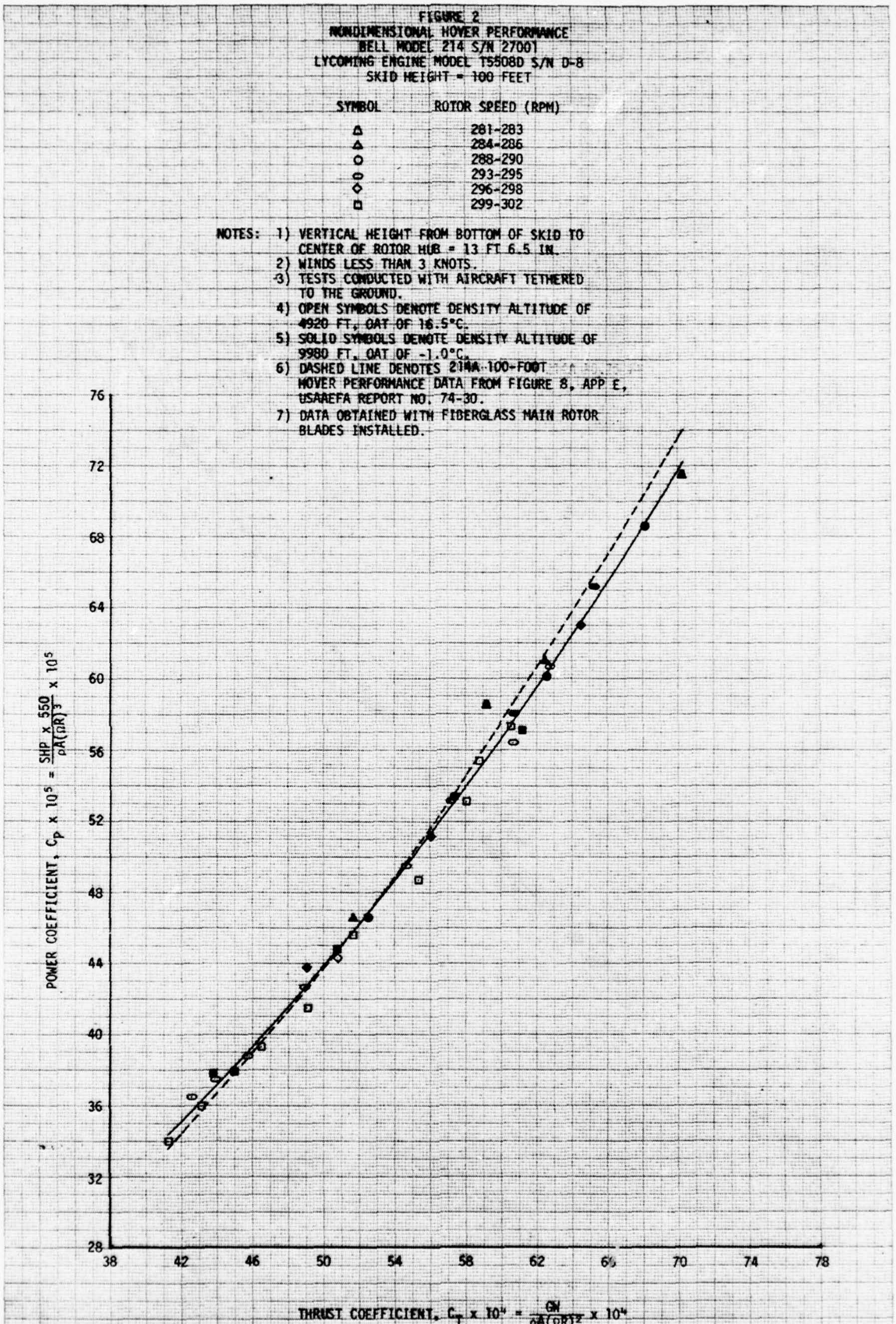


FIGURE 2
NONDIMENSIONAL HOVER PERFORMANCE
BELL MODEL 214 S/N 27001
LYCOMING ENGINE MODEL T5508D S/N D-8
SKID HEIGHT = 100 FEET
SYMBOL
ROTOR SPEED (RPM)
281-283
284-286
288-290
293-295
296-298
299-302
NOTES: 1) VERTICAL HEIGHT FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB = 13 FT 6.5 IN.
2) WINDS LESS THAN 3 KNOTS.
3) TESTS CONDUCTED WITH AIRCRAFT TETHERED TO THE GROUND.
4) OPEN SYMBOLS DENOTE DENSITY ALTITUDE OF 4920 FT, OAT OF 16.5°C.
5) SOLID SYMBOLS DENOTE DENSITY ALTITUDE OF 9980 FT, OAT OF -1.0°C.
6) DASHED LINE DENOTES 214A 100-FOOT HOVER PERFORMANCE DATA FROM FIGURE 8, APP E, USAAEFA REPORT NO. 74-30.
7) DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.
POWER COEFFICIENT, $C_P \times 10^5 = \frac{SHP \times 550}{\rho A(\Omega R)^3} \times 10^5$
THRUST COEFFICIENT, $C_T \times 10^4 = \frac{GW}{\rho A(\Omega R)^2} \times 10^4$
76
72
68
64
60
56
52
48
44
40
36
32
28
38
42
46
50
54
58
62
66
70
74
78

FIGURE 3
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
BELL MODEL 214
S/N 27001

NOTES: 1) ROTOR SPEED = 300 RPM.
2) FORWARD CENTER-OF-GRAVITY LOCATION.
3) CURVES BASED ON DATA IN FIGURES 5 THROUGH 9.
4) DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.

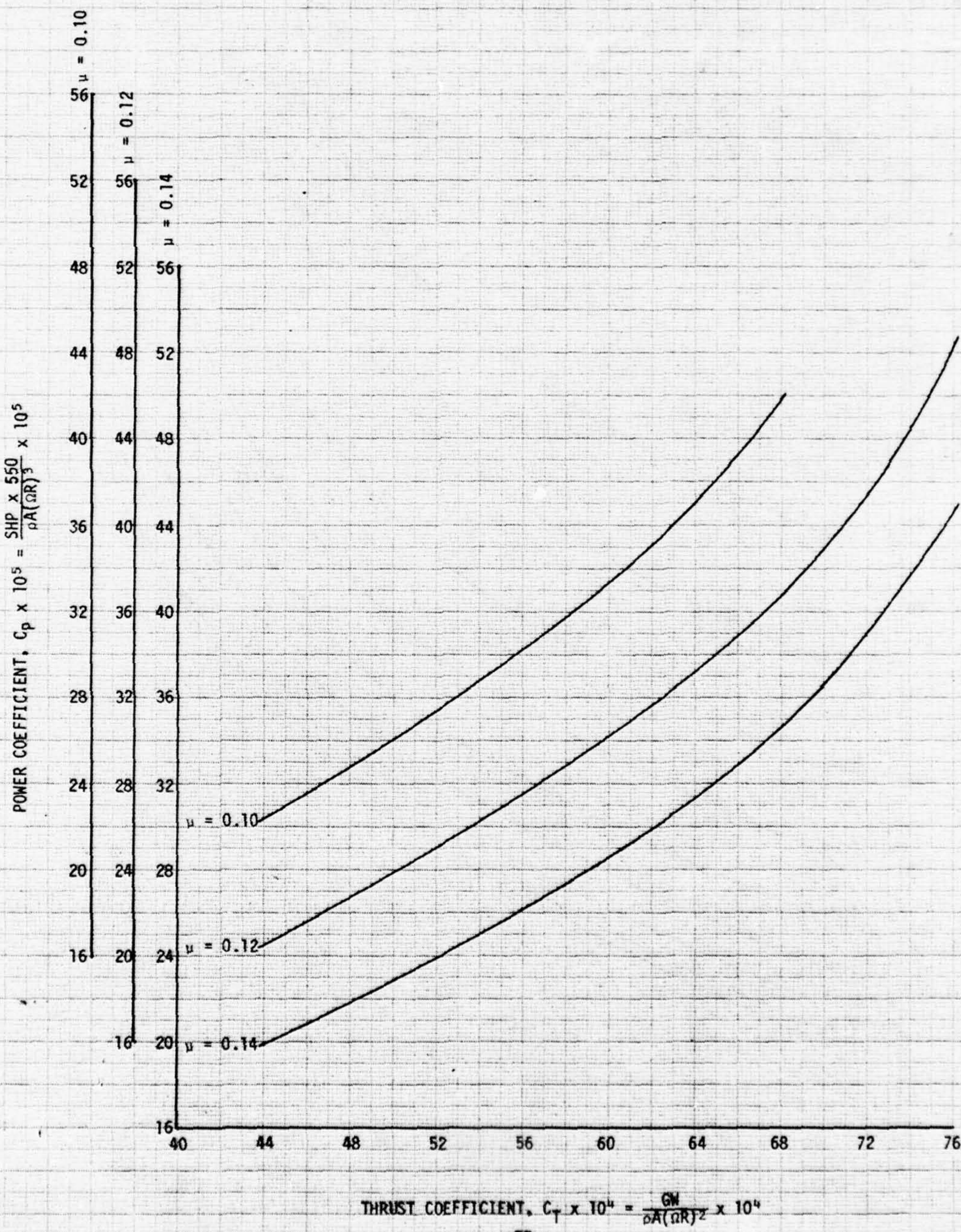


FIGURE 4
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
BELL MODEL 214
S/N 27001

NOTES: 1) ROTOR SPEED = 300 RPM.
2) FORWARD CENTER-OF-GRAVITY LOCATION.
3) CURVES BASED ON DATA IN FIGURES 5 THROUGH 9.
4) DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.

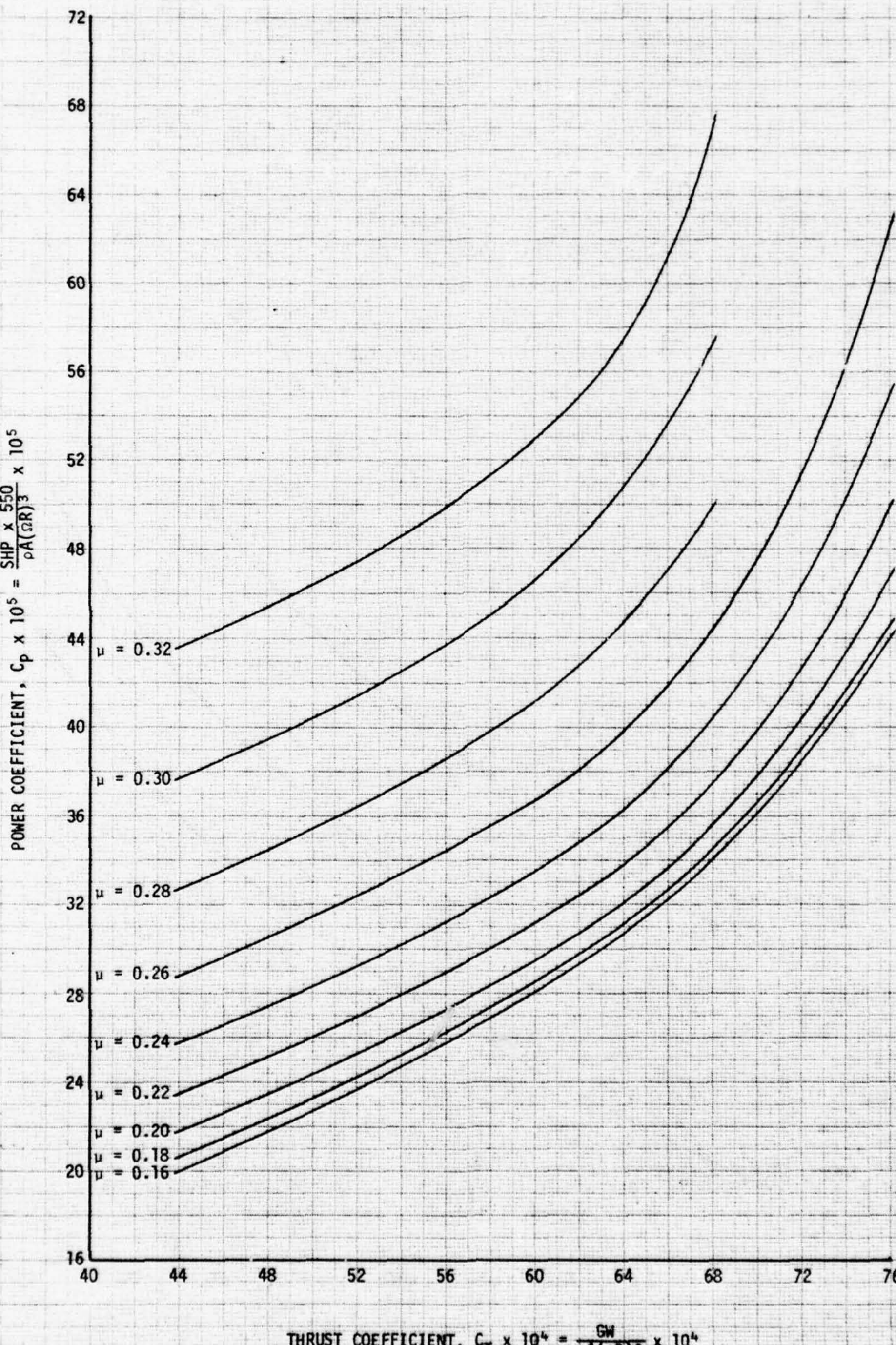


FIGURE 5
LEVEL FLIGHT PERFORMANCE
BELL MODEL 214
LYCOMING ENGINE MODEL T5508D S/N D-8

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T
10380	133.7(FWD)	0.2LT(MID)	6460	20	299	0.0044

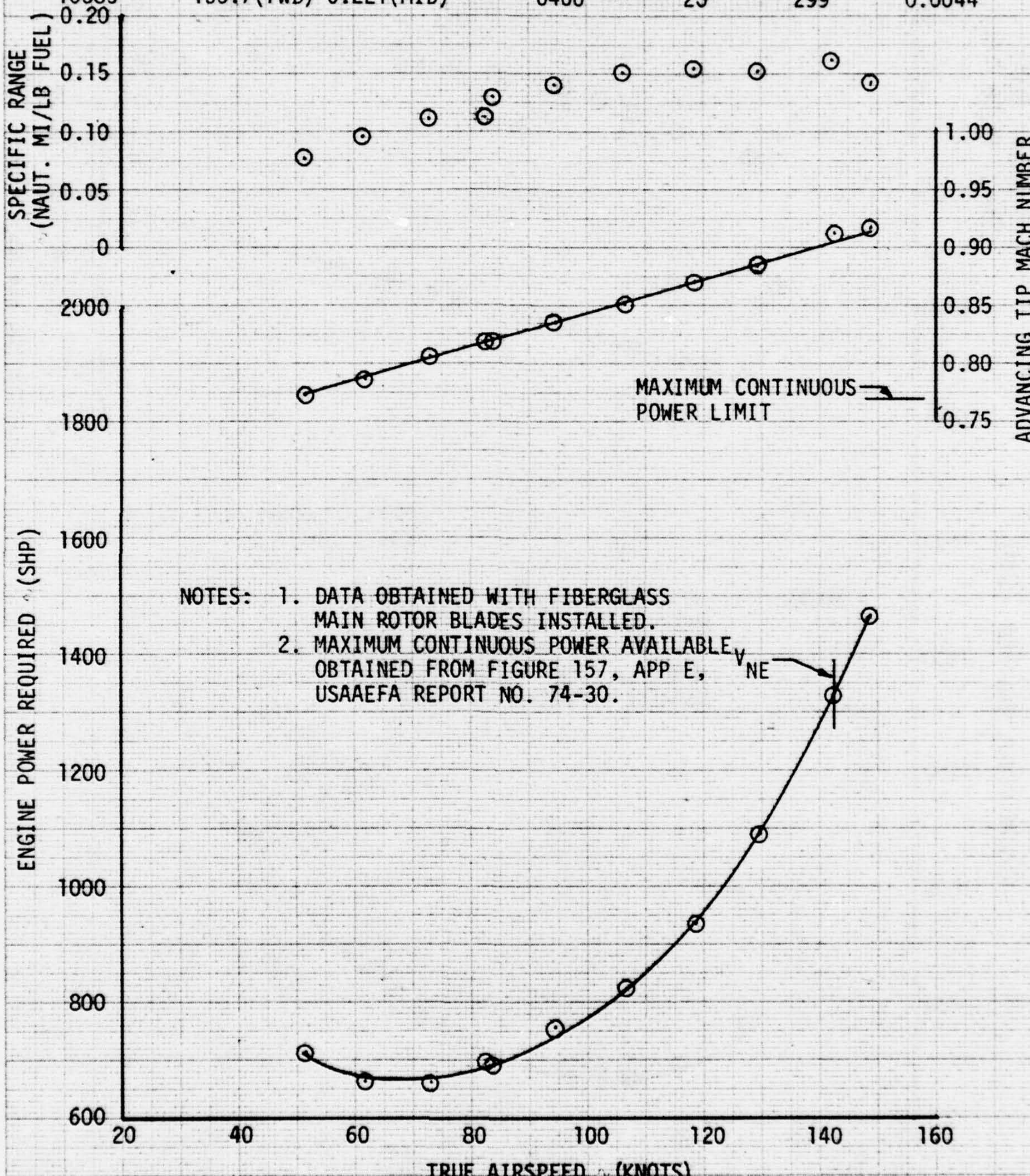


FIGURE 6
LEVEL FLIGHT PERFORMANCE
BELL MODEL 214
LYCOMING ENGINE MODEL T5508D S/N D-8

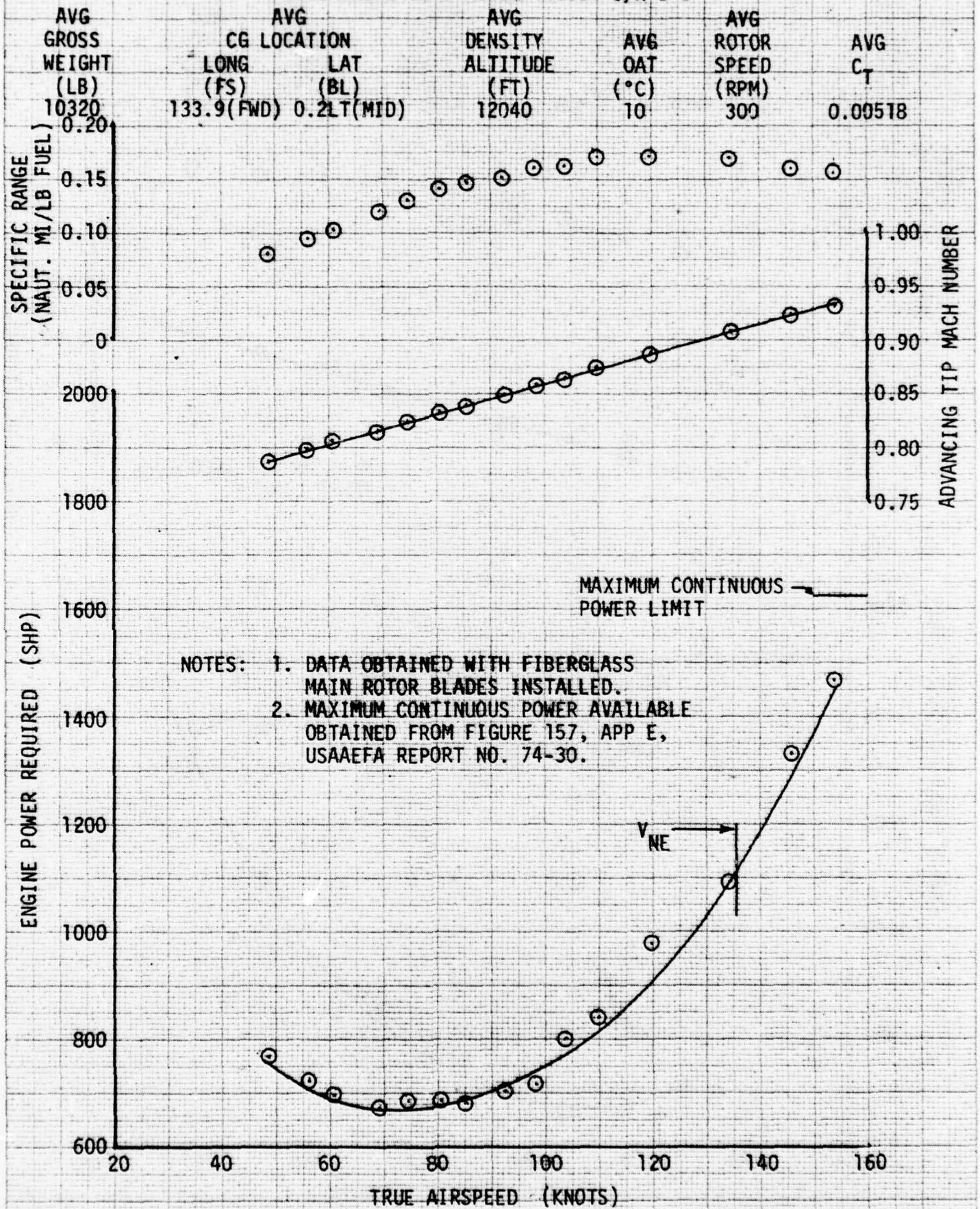


FIGURE 7
LEVEL FLIGHT PERFORMANCE
BELL MODEL 214
LYCOMING ENGINE MODEL T5508D S/N D-8

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T
13280	135.8(FWD)	0.2LT(MID)	8440	19.5	300	0.00595

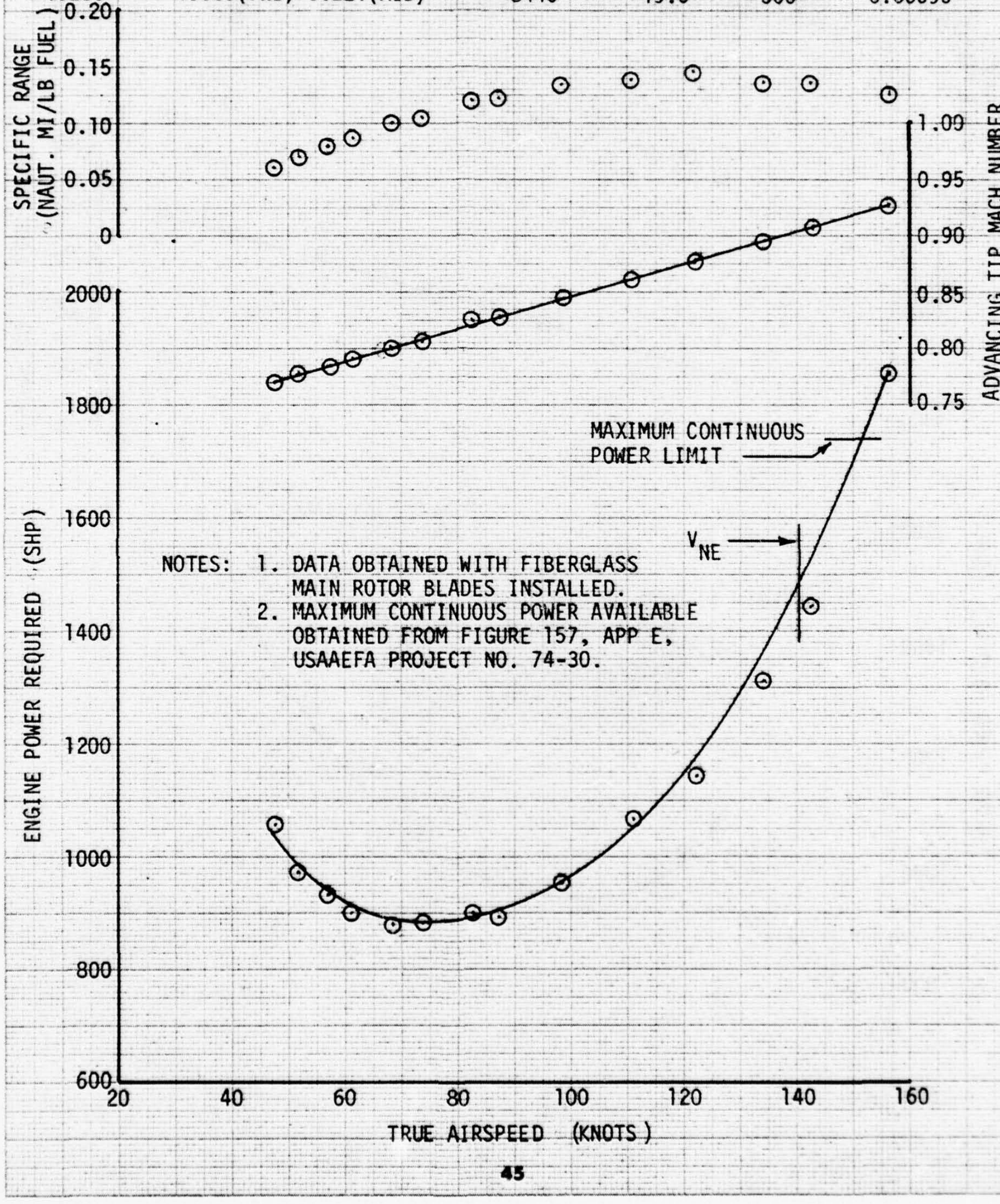


FIGURE 8
LEVEL FLIGHT PERFORMANCE
BELL MODEL 214
LYCOMING ENGINE MODEL T5508D S/N D-8

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T
13340	135.6(FWD)	0.2LT(MID)	12540	8.5	300	0.00680

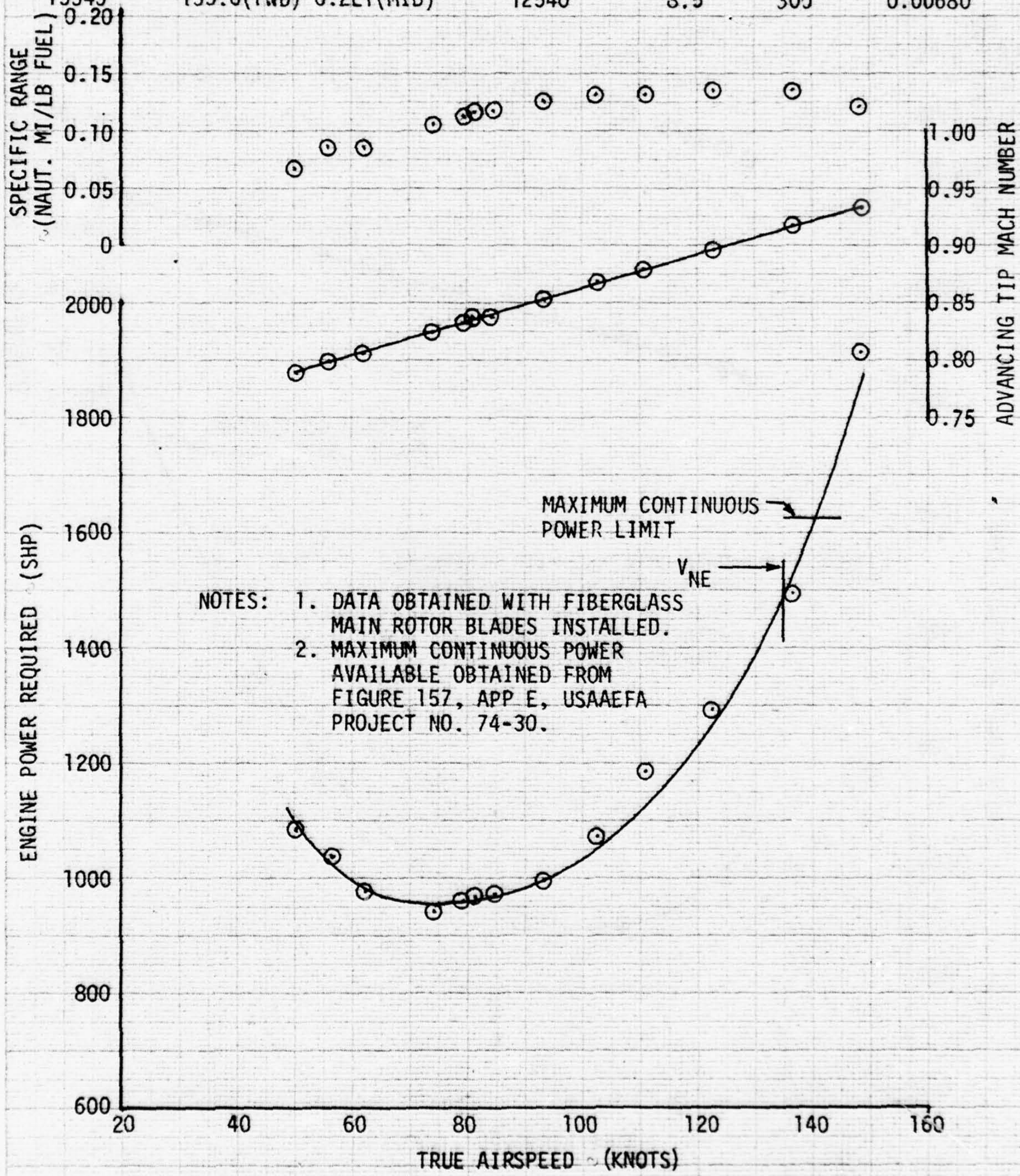


FIGURE 9
LEVEL FLIGHT PERFORMANCE
BELL MODEL 214
LYCOMING ENGINE MODEL T5508D S/N D-8

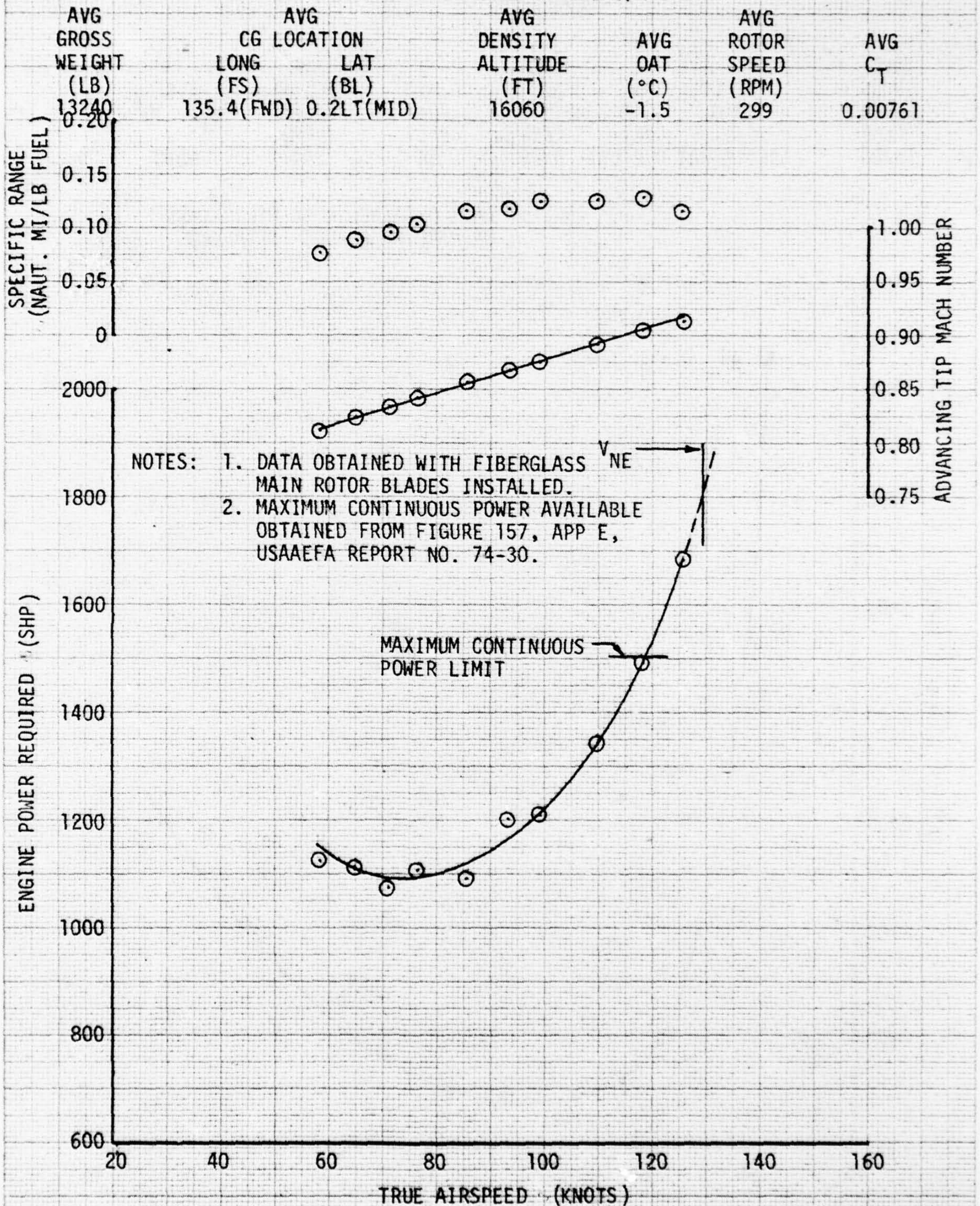


FIGURE 10
AUTOROTATIONAL DESCENT PERFORMANCE
BELL MODEL 214
S/N 27001

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG $C_T \times 10^4$
13280	136.1(FWD)	0.2(LT)	9720	14	299	62.0

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.

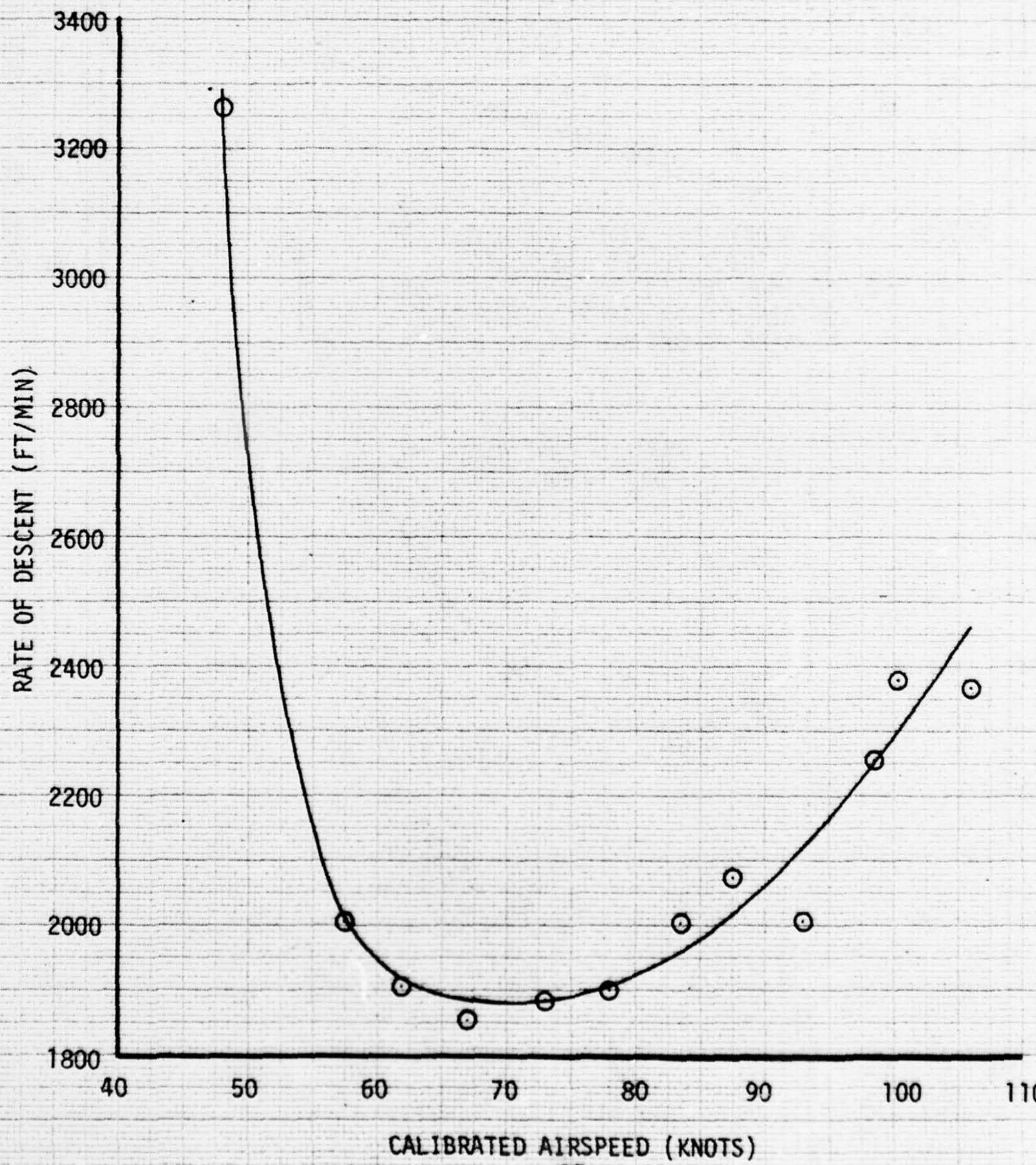


FIGURE 11
AUTOROTATIONAL DESCENT PERFORMANCE
BELL MODEL 214
S/N 27001

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG CALIB A/S (KT)	AVG $C_T \times 10^4$
13280	136.1(FWD)	0.2(LT)	9720	14	72	62.0

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.

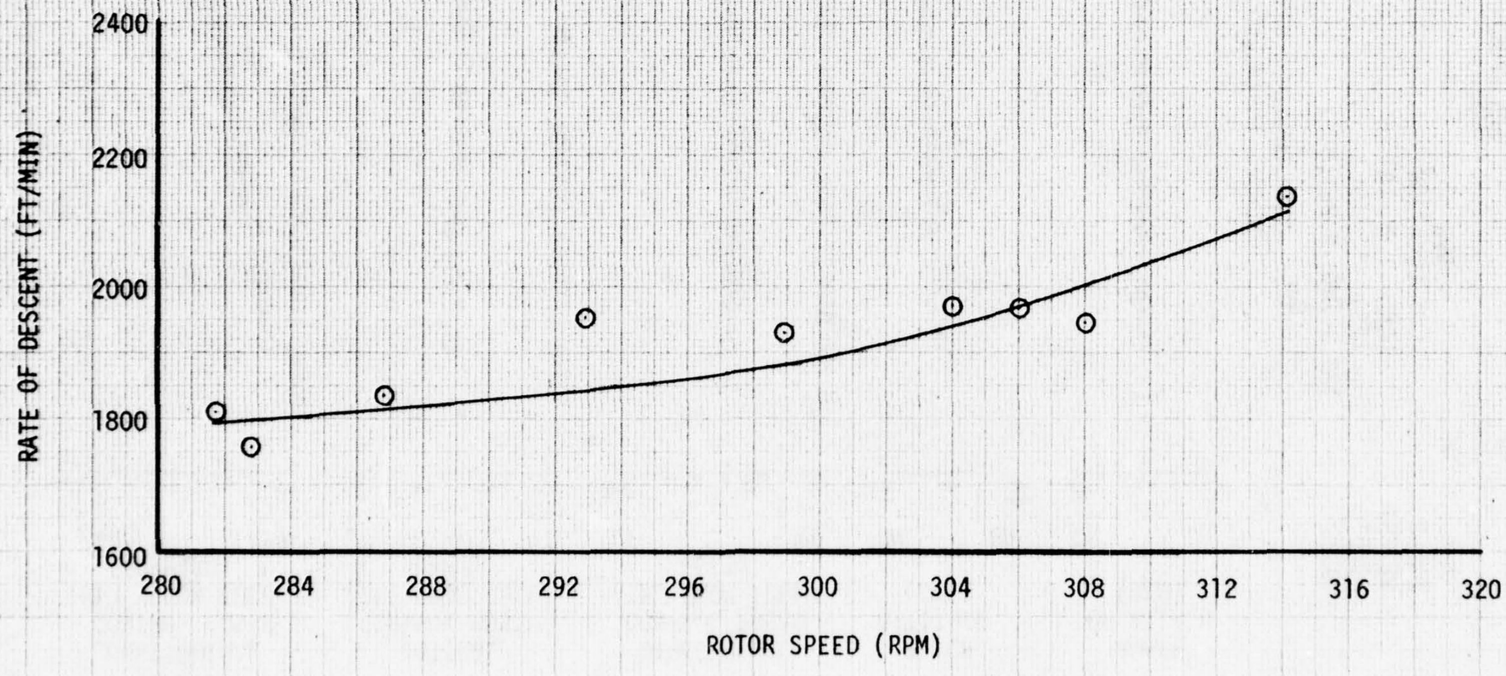


FIGURE 12
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
BELL MODEL 214
S/N 27001

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)
10320	133.9(FWD)	0.2LT(MID)	12040	10	300

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.

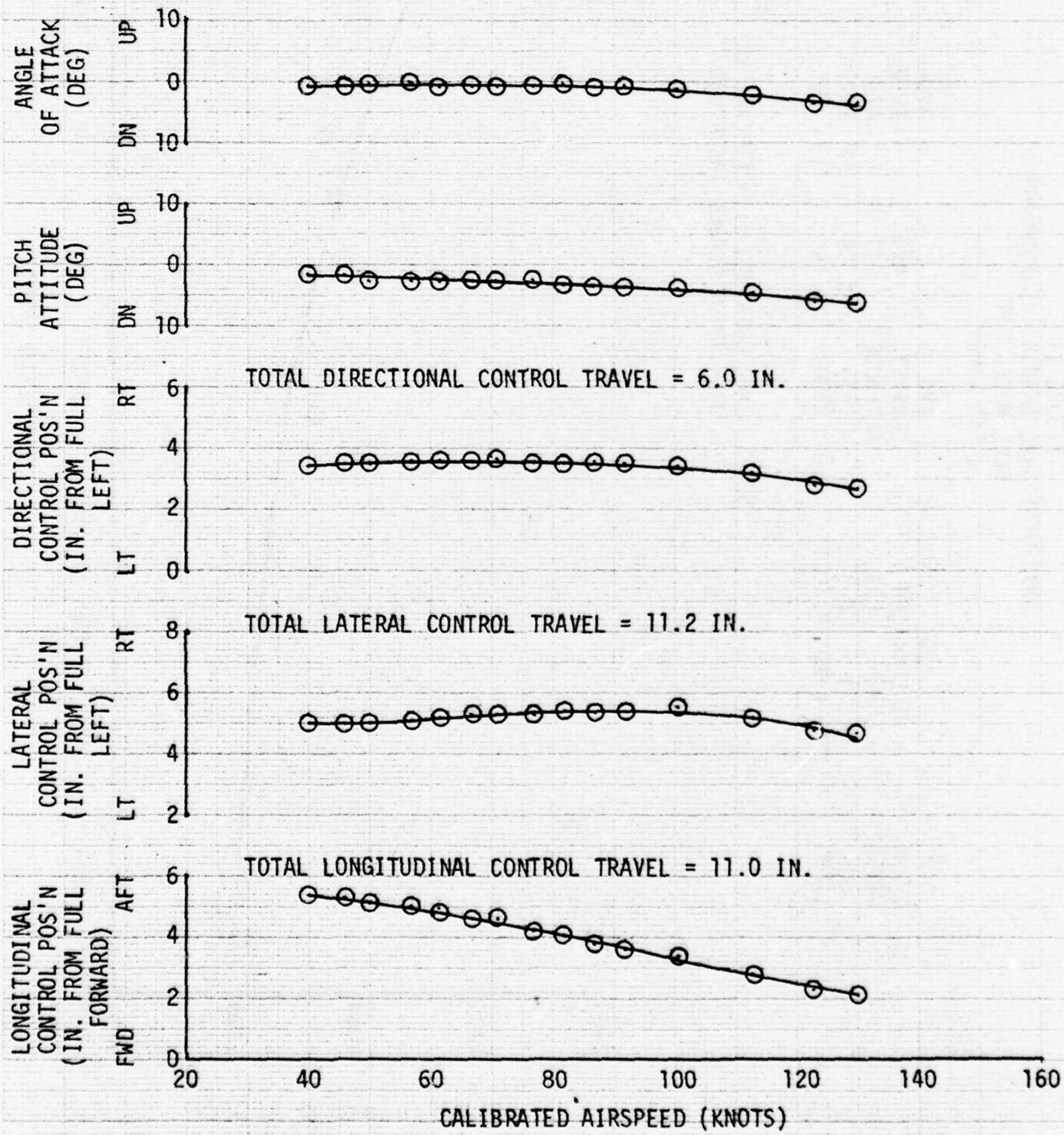


FIGURE 13
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
BELL MODEL 214
S/N 27001

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)
13280	135.8(FWD)	0.2LT(MID)	8440	19.5	300

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.

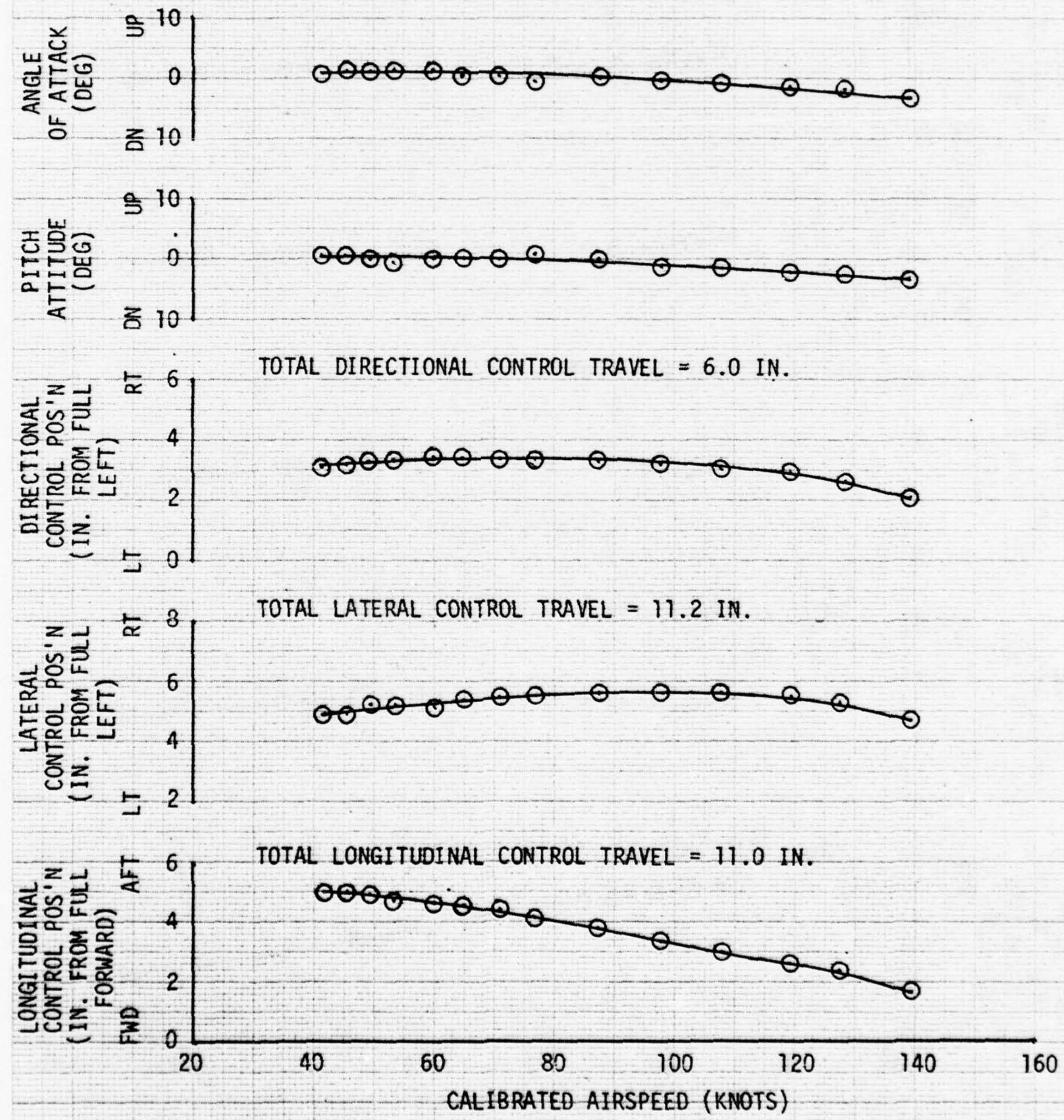


FIGURE 14
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
BELL MODEL 214
S/N 27001

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)
13340	135.6(FWD)	0.2LT(MID)	12540	8.5	300

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.

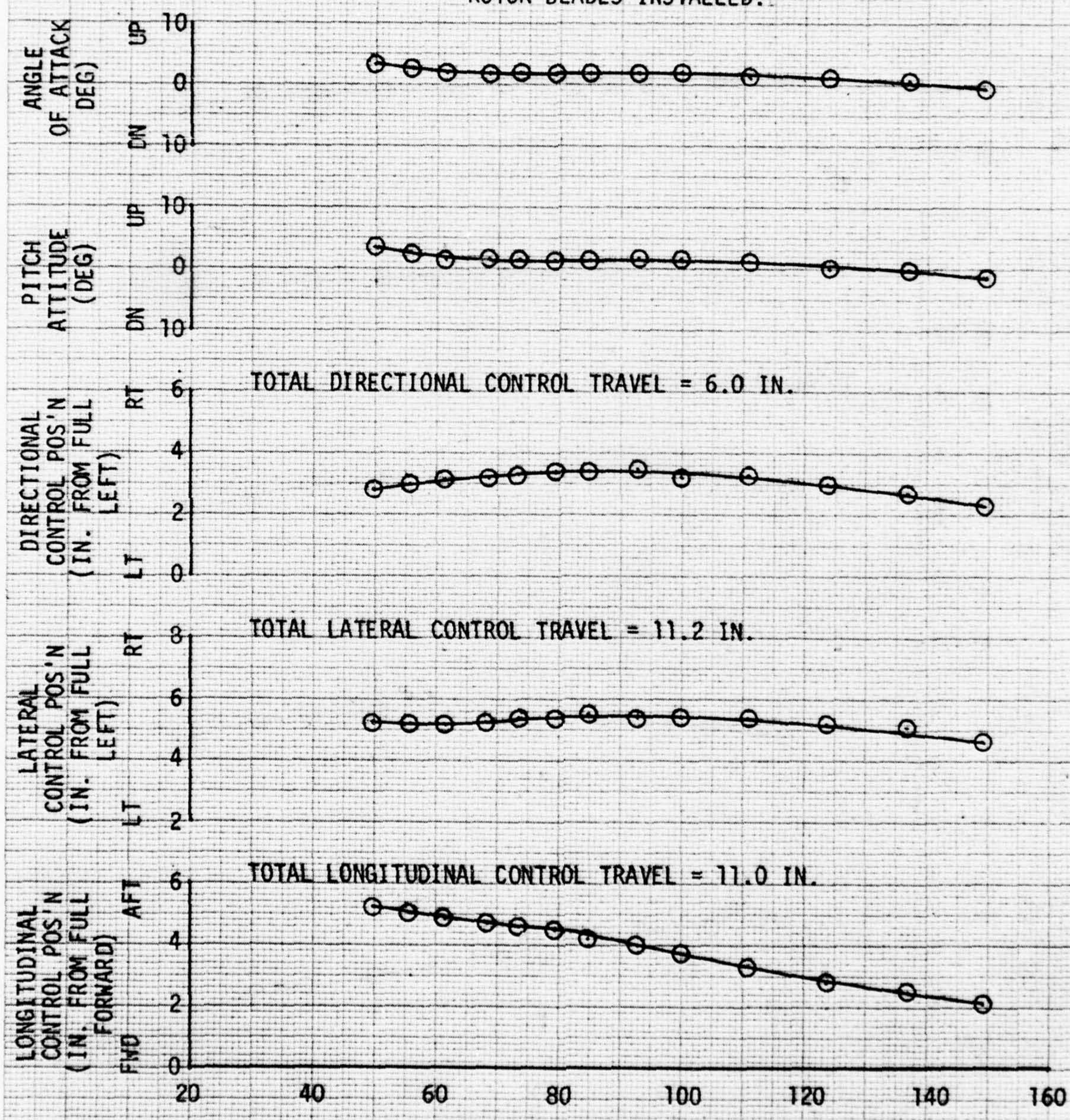


FIGURE 15
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
BELL MODEL 214
S/N 27001

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)
13240	135.4(FWD)	0.2LT(MID)	16060	-1.5	299

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.

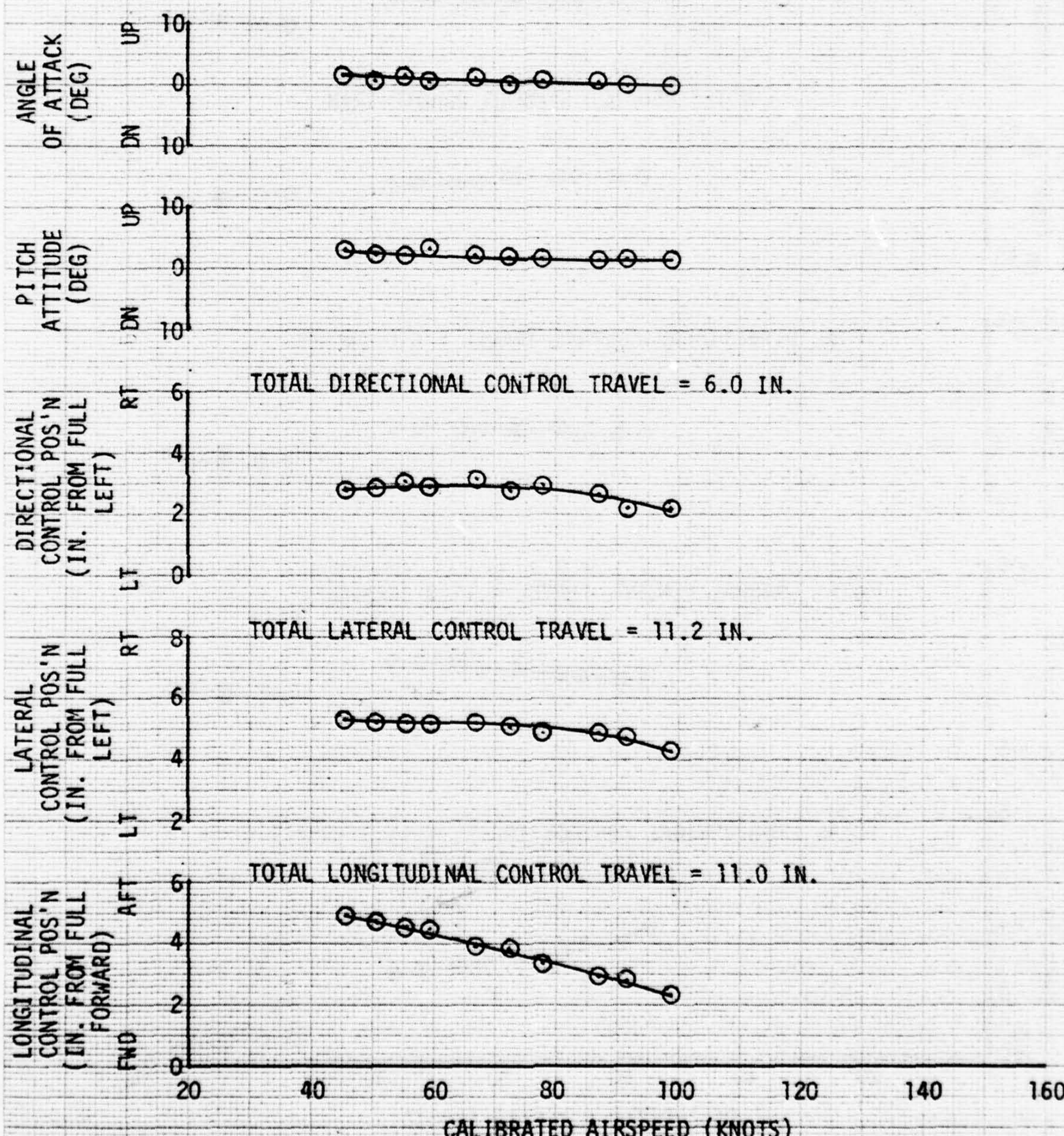


FIGURE 16
STATIC LONGITUDINAL STABILITY
BELL MODEL 214
S/N 27001

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
13840	140.3(MID)	0.2LT(MID)	4780	20	300	LEVEL FLT

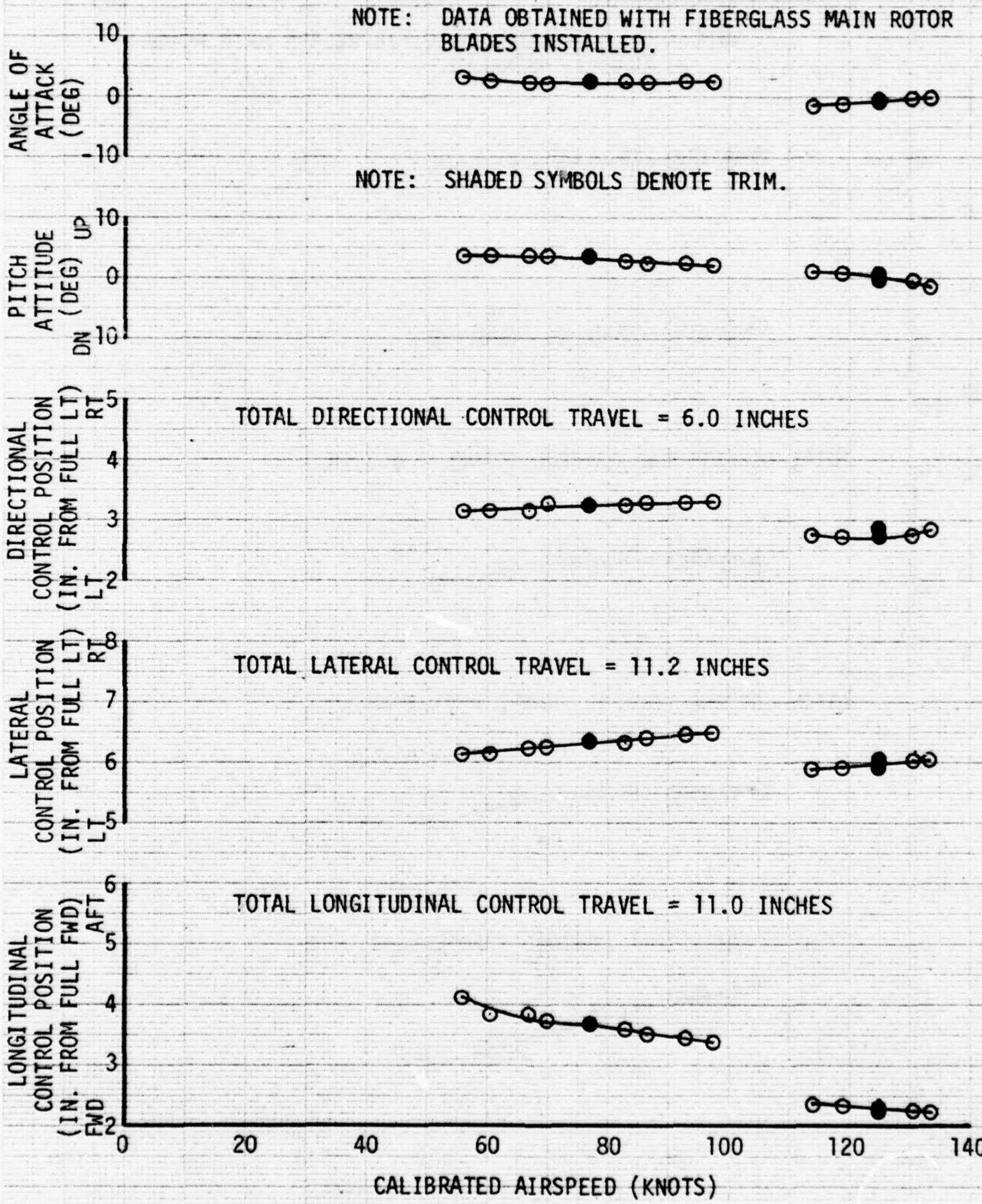


FIGURE 17
STATIC LATERAL-DIRECTIONAL STABILITY
BELL MODEL 214
S/N 27001

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)
○	13780	139.9(MID)	0.1LT(MID)	5080	22	300	37
□	13480	139.2(MID)	0.1LT(MID)	5300	24	300	125

NOTE: 1. SHADED SYMBOLS DENOTE TRIM IN LEVEL FLIGHT
2. DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.

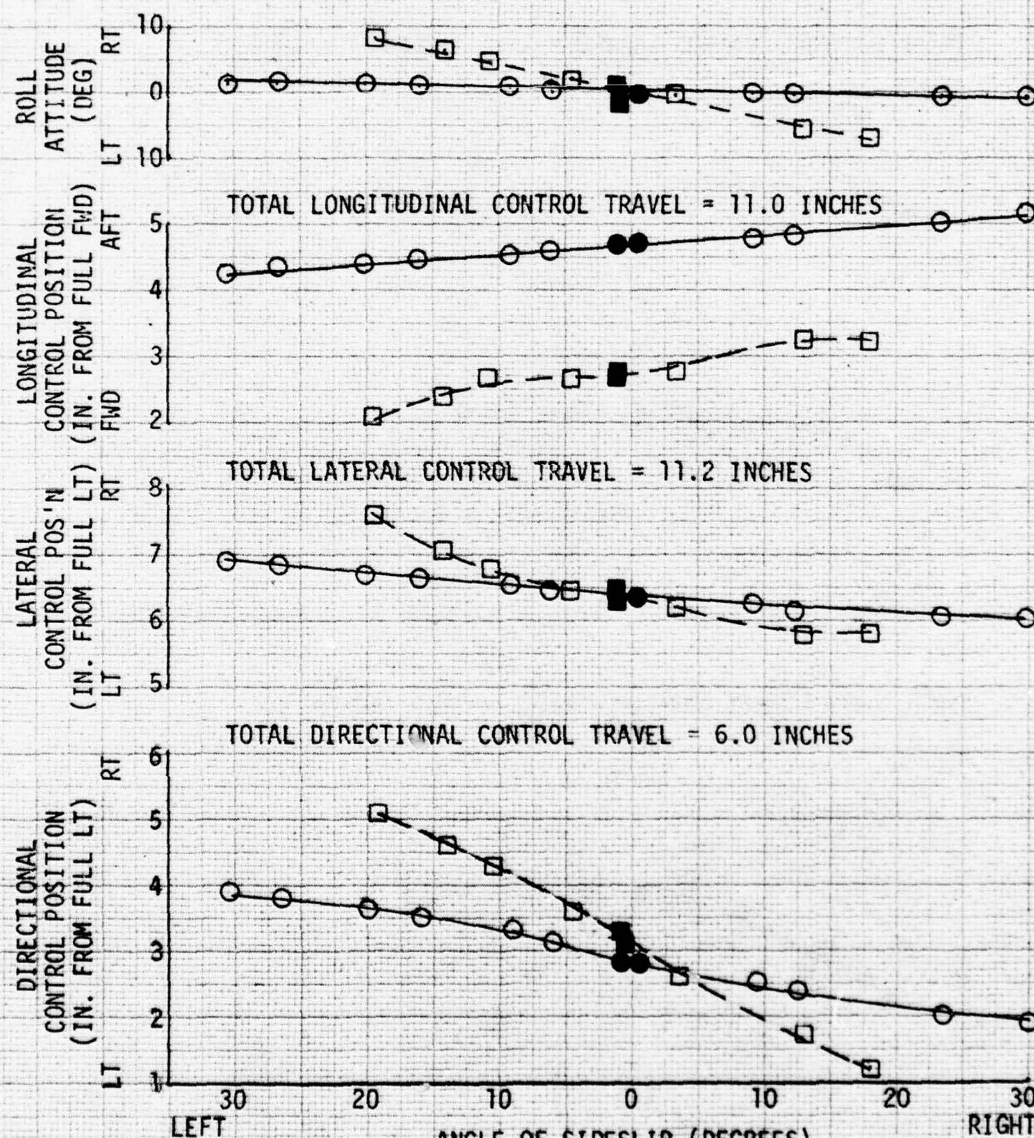


FIGURE 18
MANEUVERING STABILITY
BELL MODEL 214
S/N 27001

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	FLIGHT CONDITION
13340	140.3(AFT)	0.2LT(MID)	5100	22.0	300	126	RIGHT TURN

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.

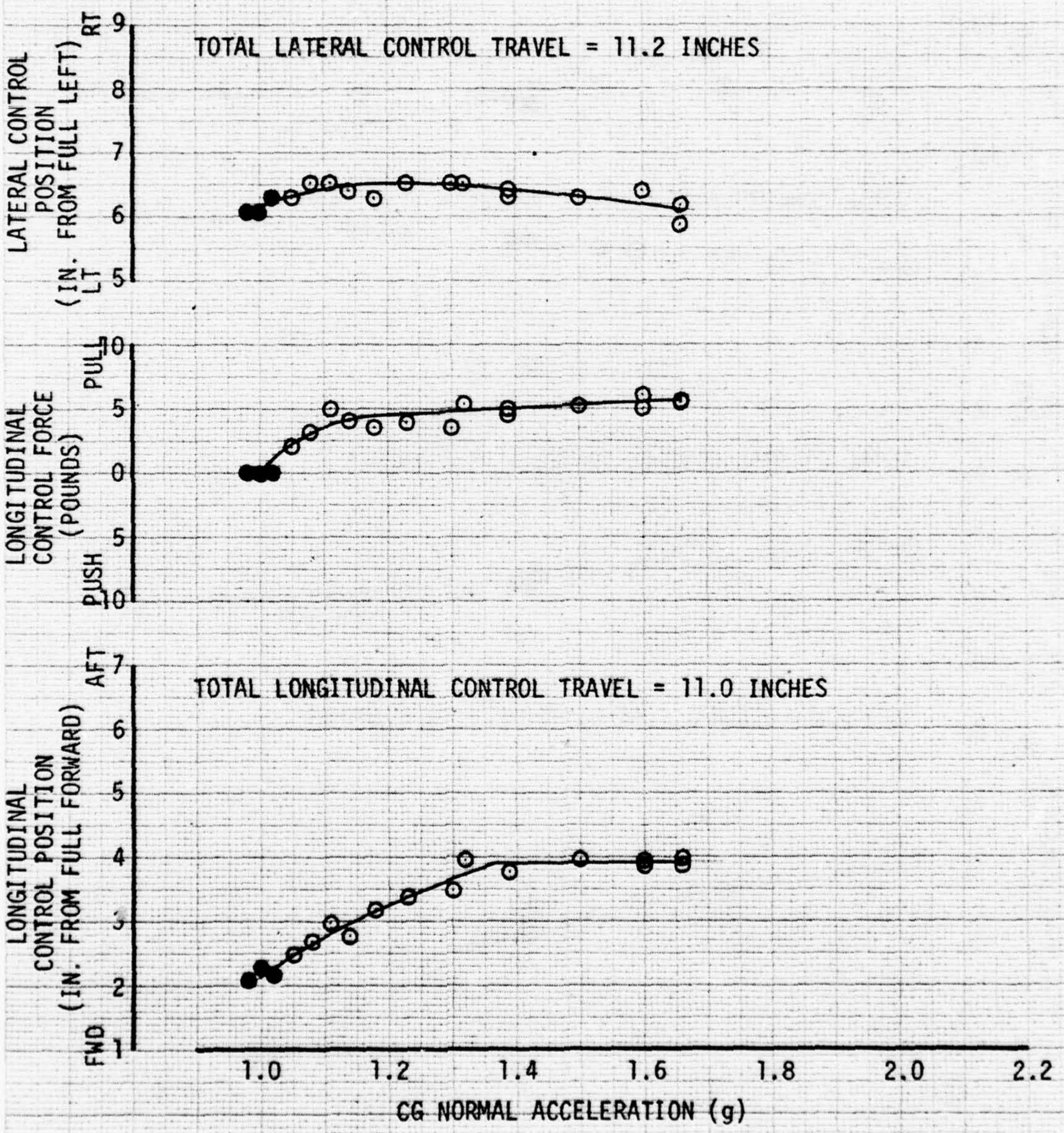


FIGURE 19
DIRECTIONAL PULSE
BELL MODEL 214
S/N 27001

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)
13340	139.9(AFT)	0.2LT(MID)	2940	26.5	303	0

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.
TOTAL CONTROL TRAVEL: 11.0 INCHES LONGITUDINAL; 11.2 INCHES LATERAL; 6.0 DIRECTIONAL.

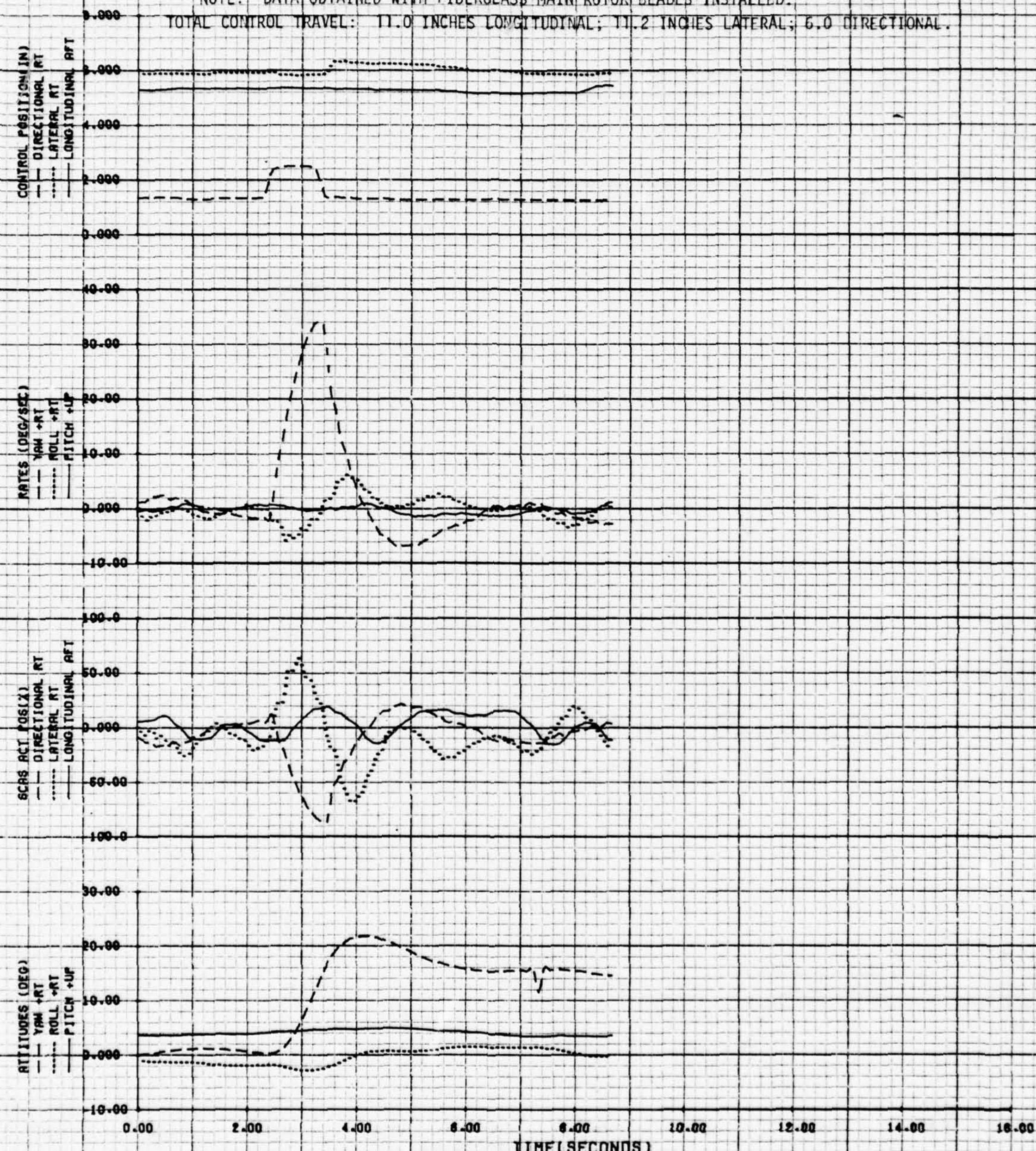
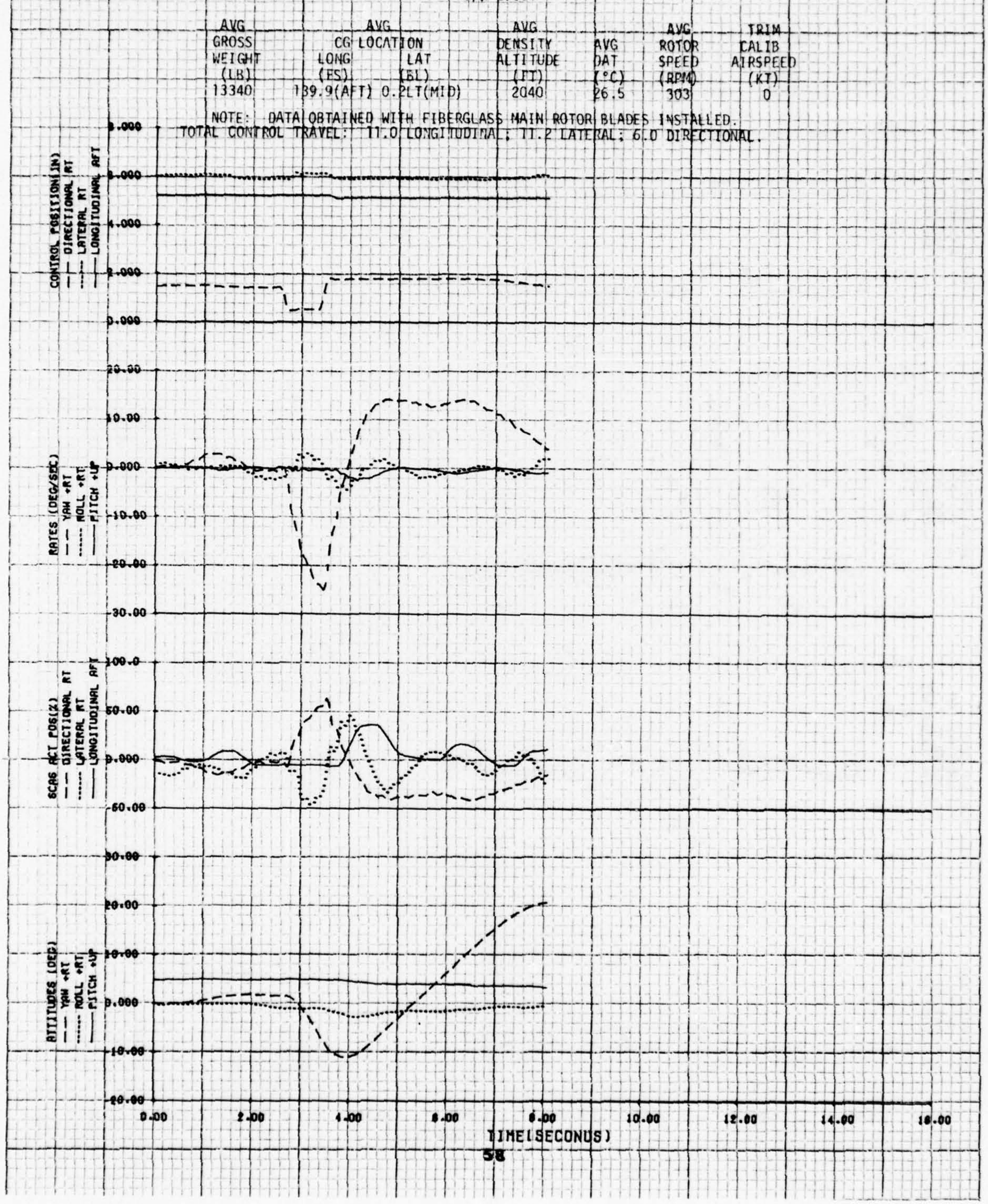


FIGURE 20
DIRECTIONAL PULSE
BELL MODEL 214
S/N 27001



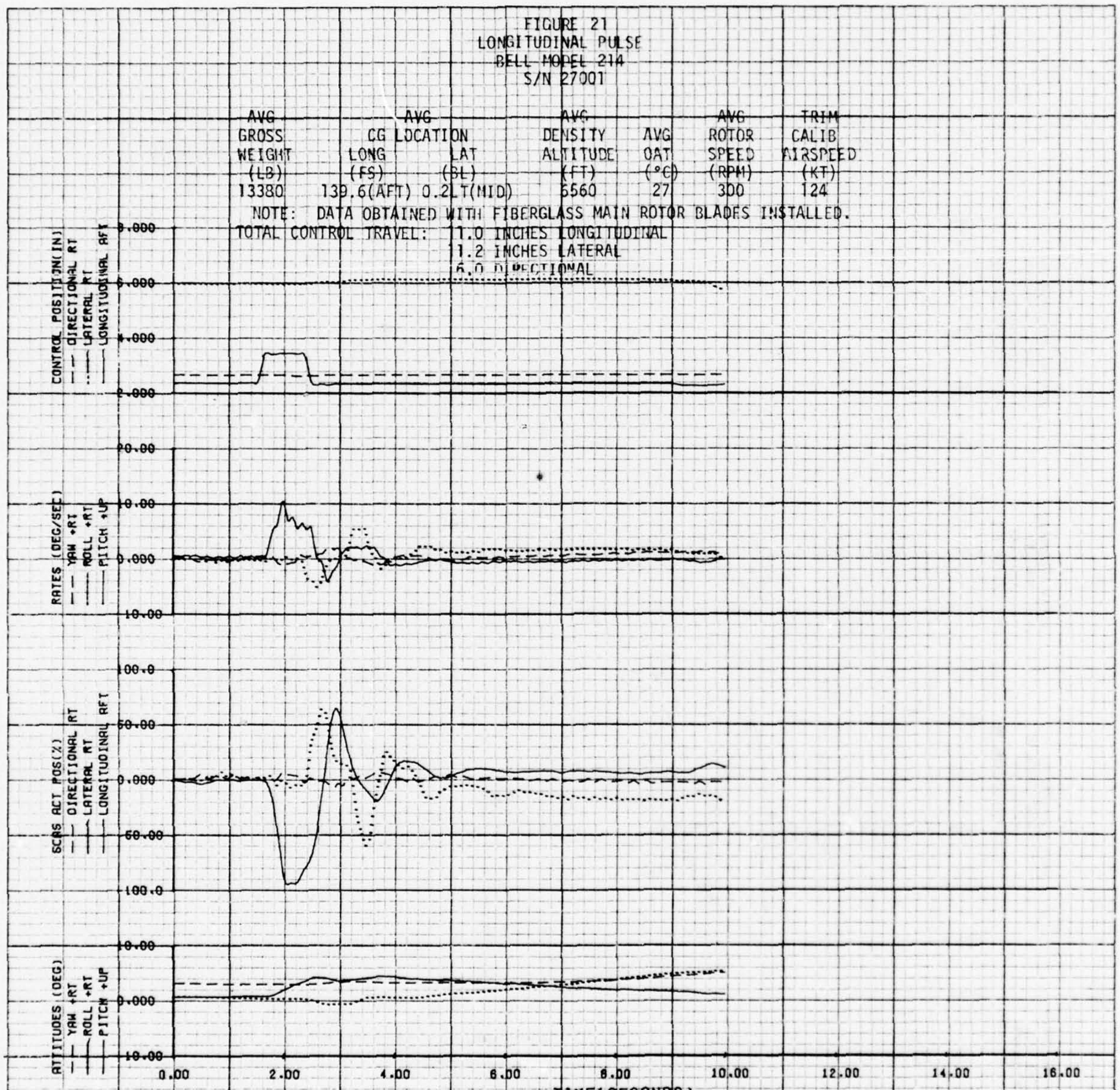


FIGURE 21
LONGITUDINAL PULSE
BELL MODEL 214
S/N 27001
AVG GROSS WEIGHT (LB) 13380
AVG CG LOCATION LONG (FS) 139.6(AFT) LAT (BL) 0.2LT(MID)
AVG DENSITY ALTITUDE (FT) 5560
AVG OAT (°C) 27
AVG ROTOR SPEED (RPM) 300
TRIM CALIB AIRSPEED (KT) 124
NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.
TOTAL CONTROL TRAVEL: 11.0 INCHES LONGITUDINAL
11.2 INCHES LATERAL
6.0 DIRECTIONAL
CONTROL POSITION(IN)
DIRECTIONAL RT
LATERAL RT
LONGITUDINAL AFT
RATES (DEG/SEC)
YAW +RT
ROLL +RT
PITCH +UP
SCAS ACT POS(%)
DIRECTIONAL RT
LATERAL RT
LONGITUDINAL AFT
ATTITUDES (DEG)
YAW +RT
ROLL +RT
PITCH +UP
0.00
2.00
4.00
6.00
8.00
10.00
12.00
14.00
16.00
TIME(SECONDS)

FIGURE 22
LONGITUDINAL PULSE
BELL MODEL 214
S/N 27001

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)
13380	139.6(AFT)	0.2LT(MID)	6560	27	300	124

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.
TOTAL CONTROL TRAVEL: 11.0 INCHES LONGITUDINAL
11.2 INCHES LATERAL
5.0 INCHES DIRECTIONAL

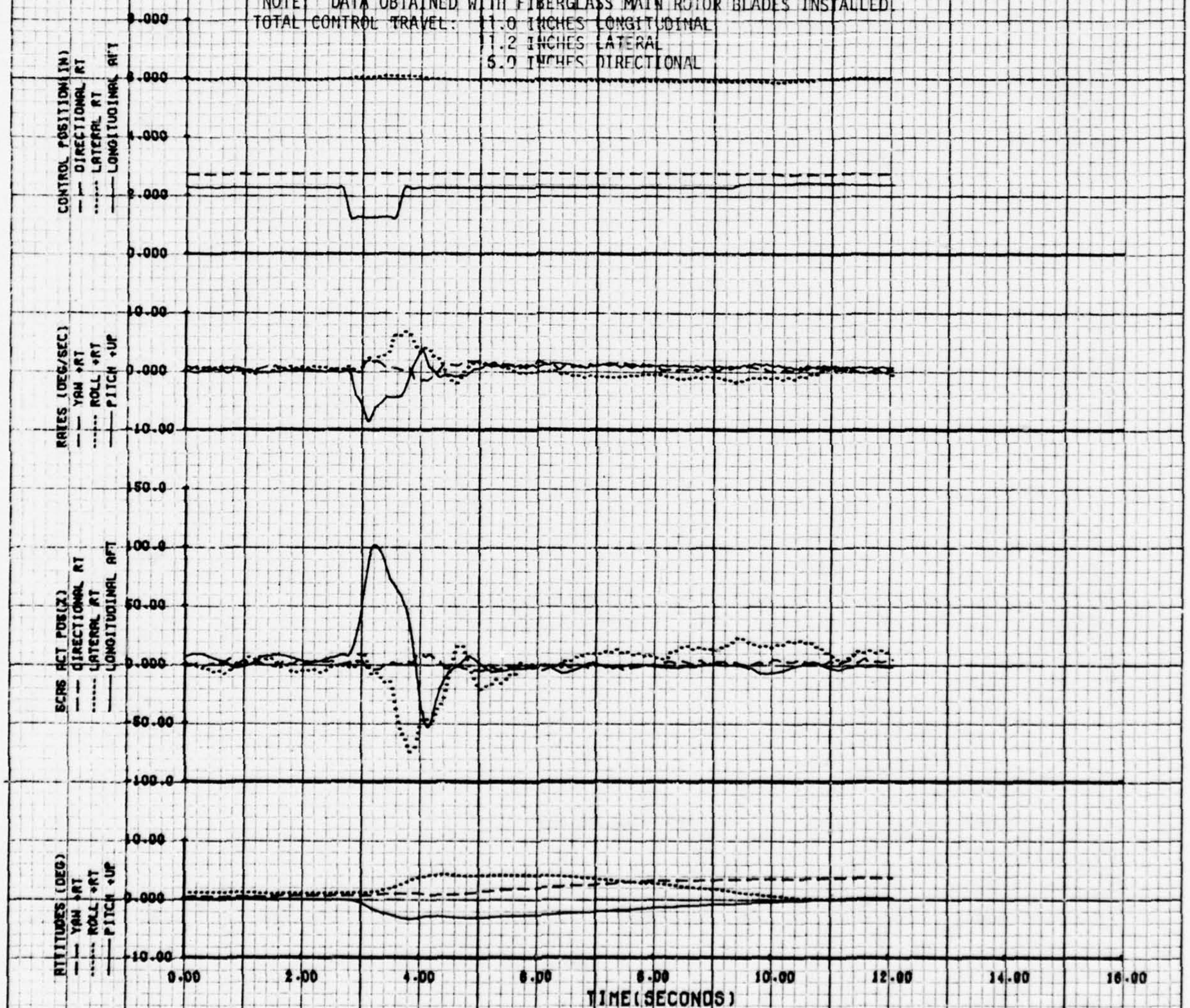
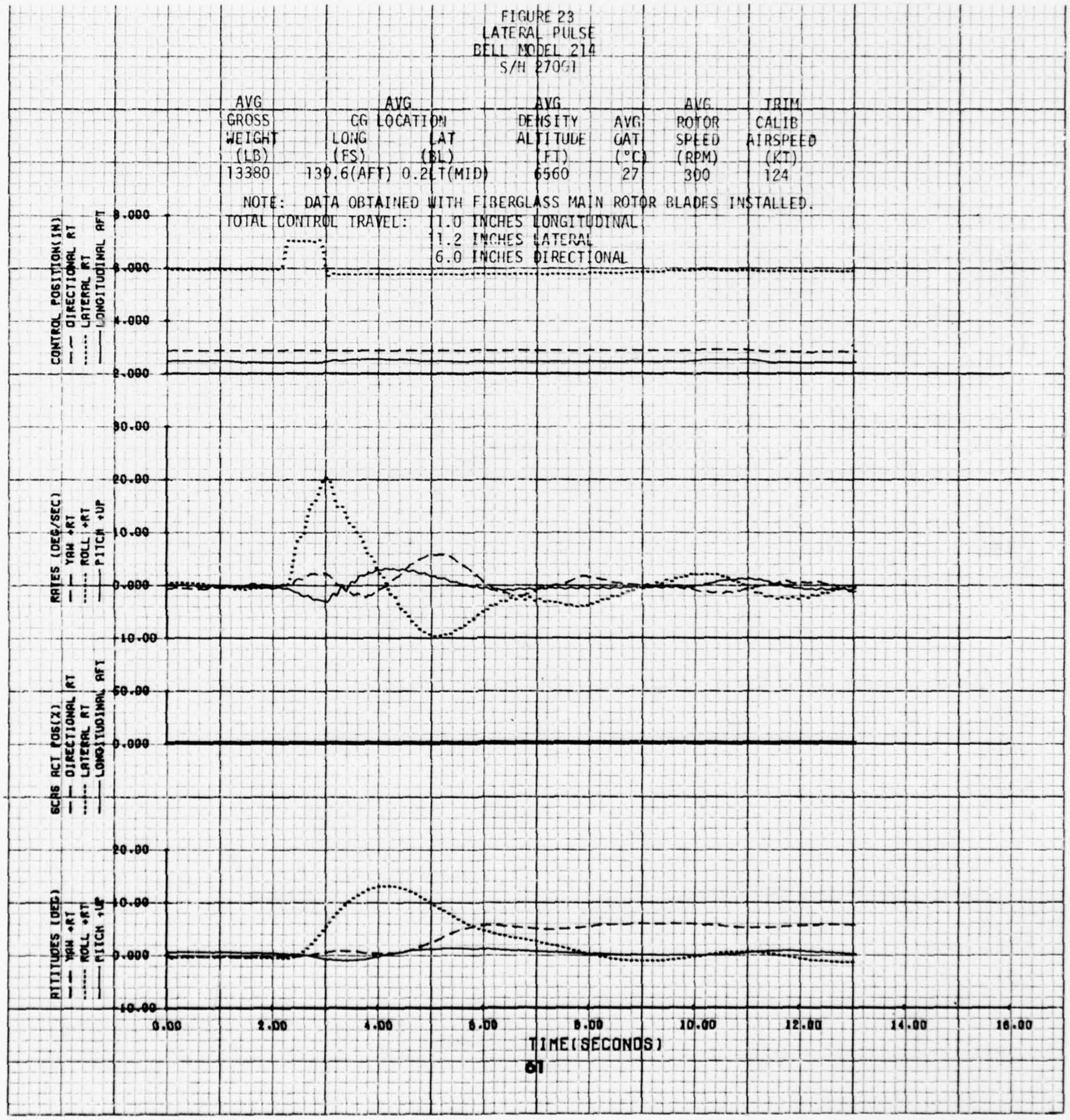


FIGURE 23
LATERAL PULSE
BELL MODEL 214
S/N 27061



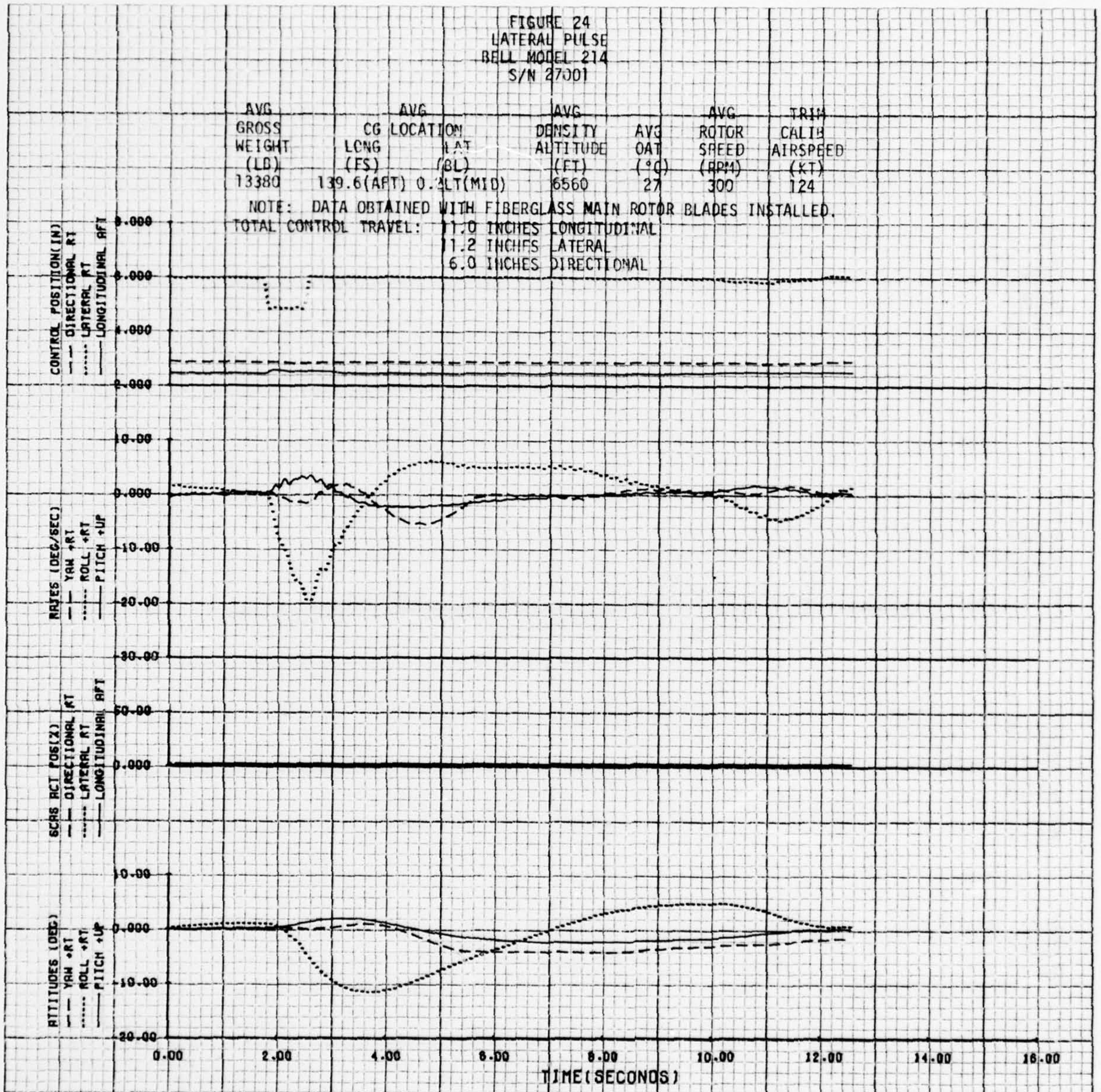


FIGURE 24
LATERAL PULSE
BELL MODEL 214
S/N 27001
AVG GROSS WEIGHT (LB) 13380
AVG CG LOCATION LONG (FS) 139.6(AFT) LAT (BL) 0.4LT(MID)
AVG DENSITY ALTITUDE (FT) 6560
AVG OAT (°C) 27
AVG ROTOR SPEED (RPM) 300
TRIM CALIB AIRSPEED (KT) 124
NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.
TOTAL CONTROL TRAVEL: 11.0 INCHES LONGITUDINAL
11.2 INCHES LATERAL
6.0 INCHES DIRECTIONAL
CONTROL POSITION(IN)
DIRECTIONAL RT
LATERAL RT
LONGITUDINAL AFT
RATES (DEG/SEC)
YAW +RT
ROLL +RT
PITCH +UP
SCAS ACT POS(%)
DIRECTIONAL RT
LATERAL RT
LONGITUDINAL AFT
ATTITUDES (DEG)
YAW +RT
ROLL +RT
PITCH +UP
TIME(SECONDS)

FIGURE 25
LONGITUDINAL CONTROLLABILITY
BELL MODEL 214
S/N 27001

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM CALIB. AIRSPEED (KT)
13340	139.9(AFT)	0.2LT(MID)	2040	26.5	303	0

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.

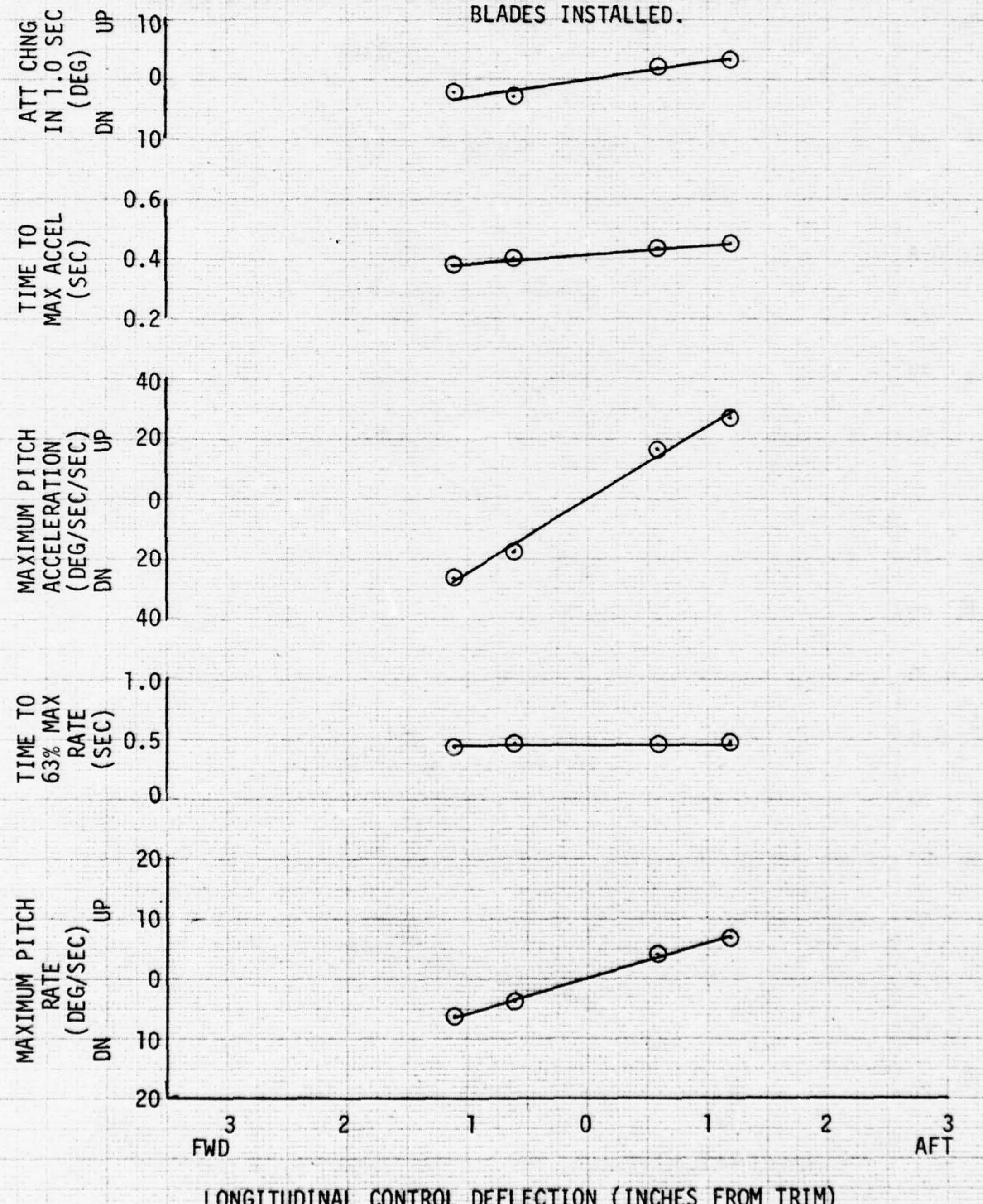


FIGURE 26
LONGITUDINAL CONTROLLABILITY
BELL MODEL 214
S/N 27001

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM CALIB. AIRSPEED (KT)
13380	139.6(AFT)	0.2LT(MID)	6560	27	300	124

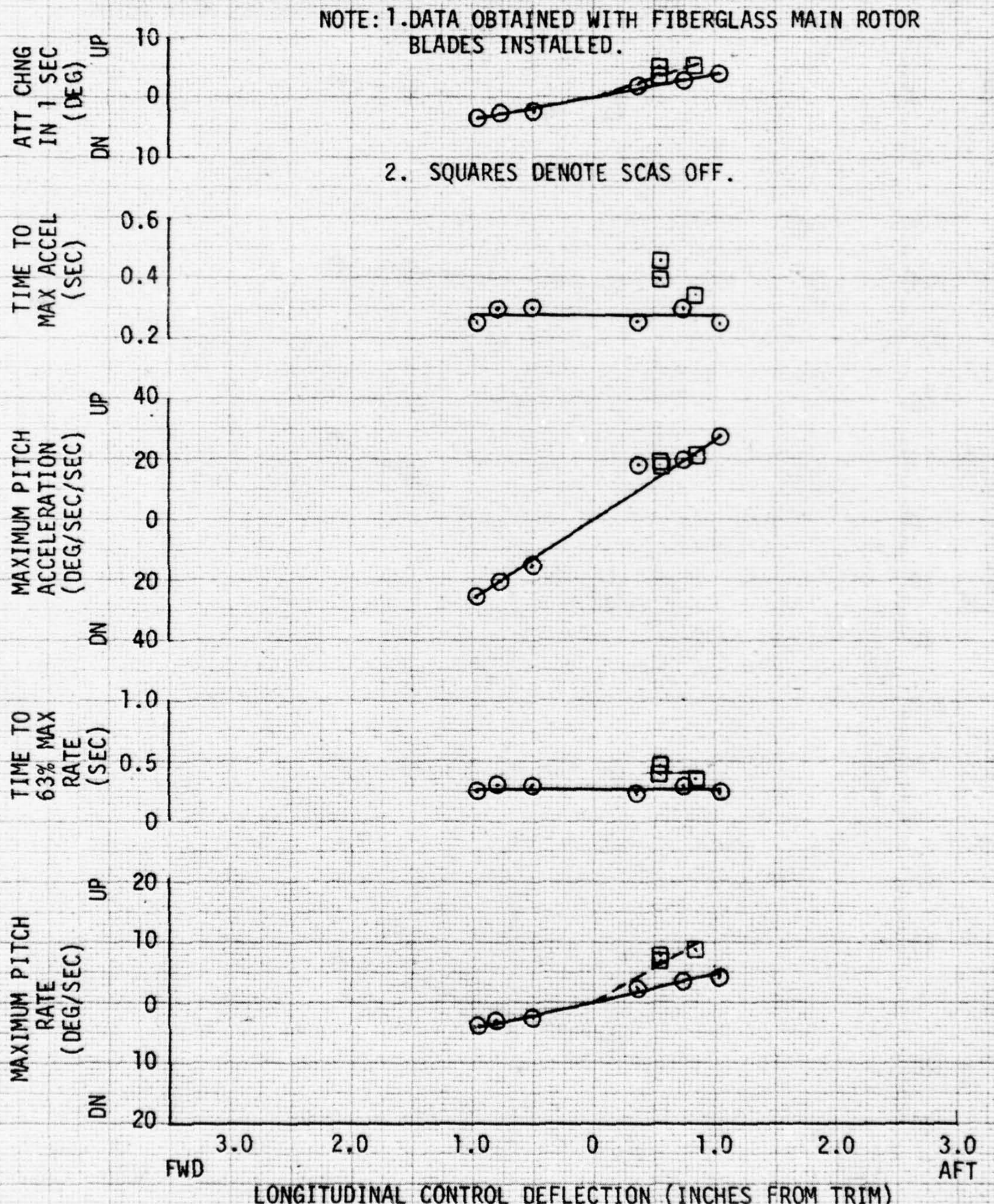


FIGURE 27
LATERAL CONTROLLABILITY
BELL MODEL 214
S/N 27001

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM CALIB. AIRSPEED (KT)
13340	139.9(AFT)	0.2LT(MID)	2040	26.5	303	0

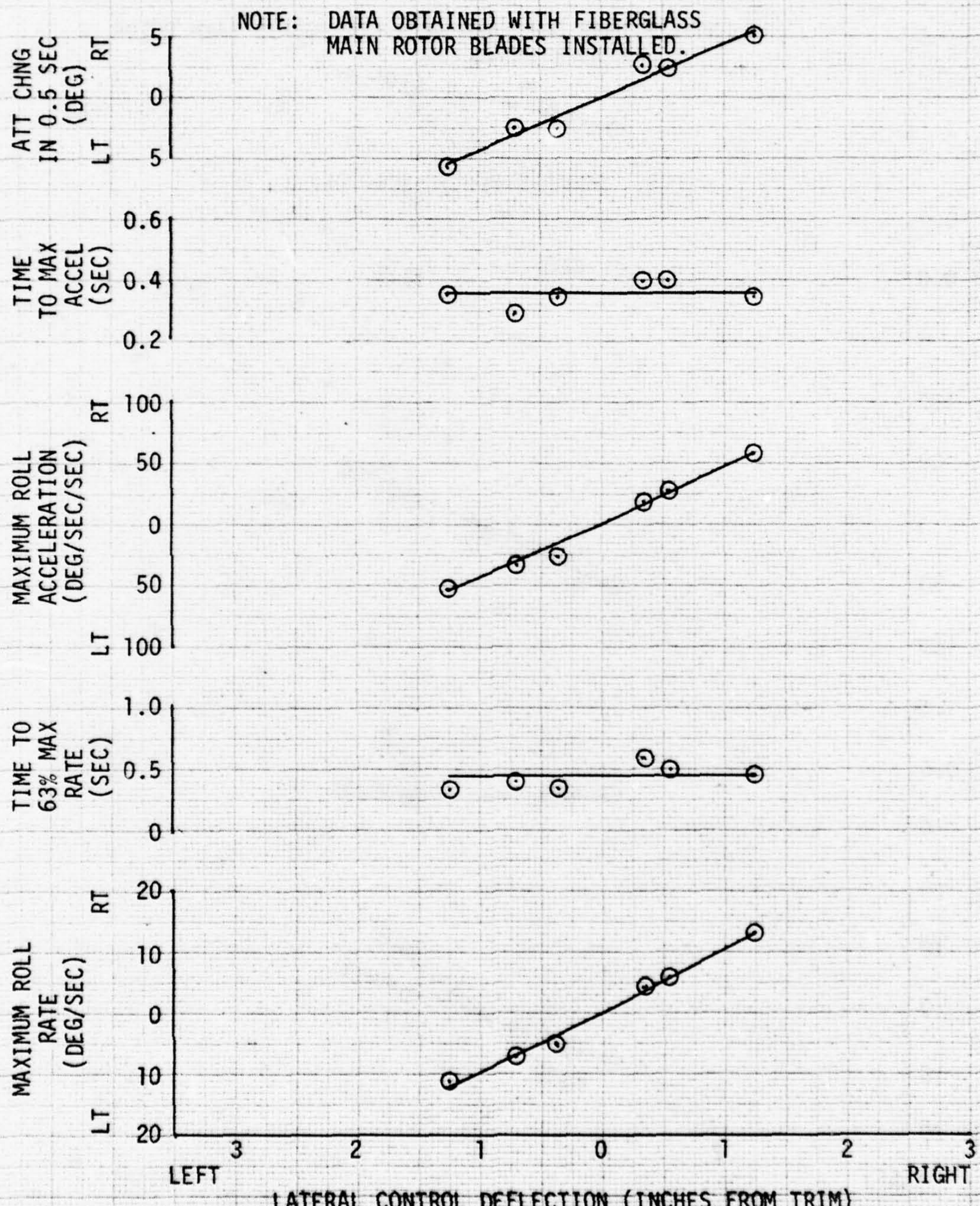


FIGURE 28
LATERAL CONTROLLABILITY
BELL MODEL 214
S/N 27001

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM CALIB. AIRSPEED (KT)
13380	139.6(AFT)	0.2LT(MID)	6560	27	300	124

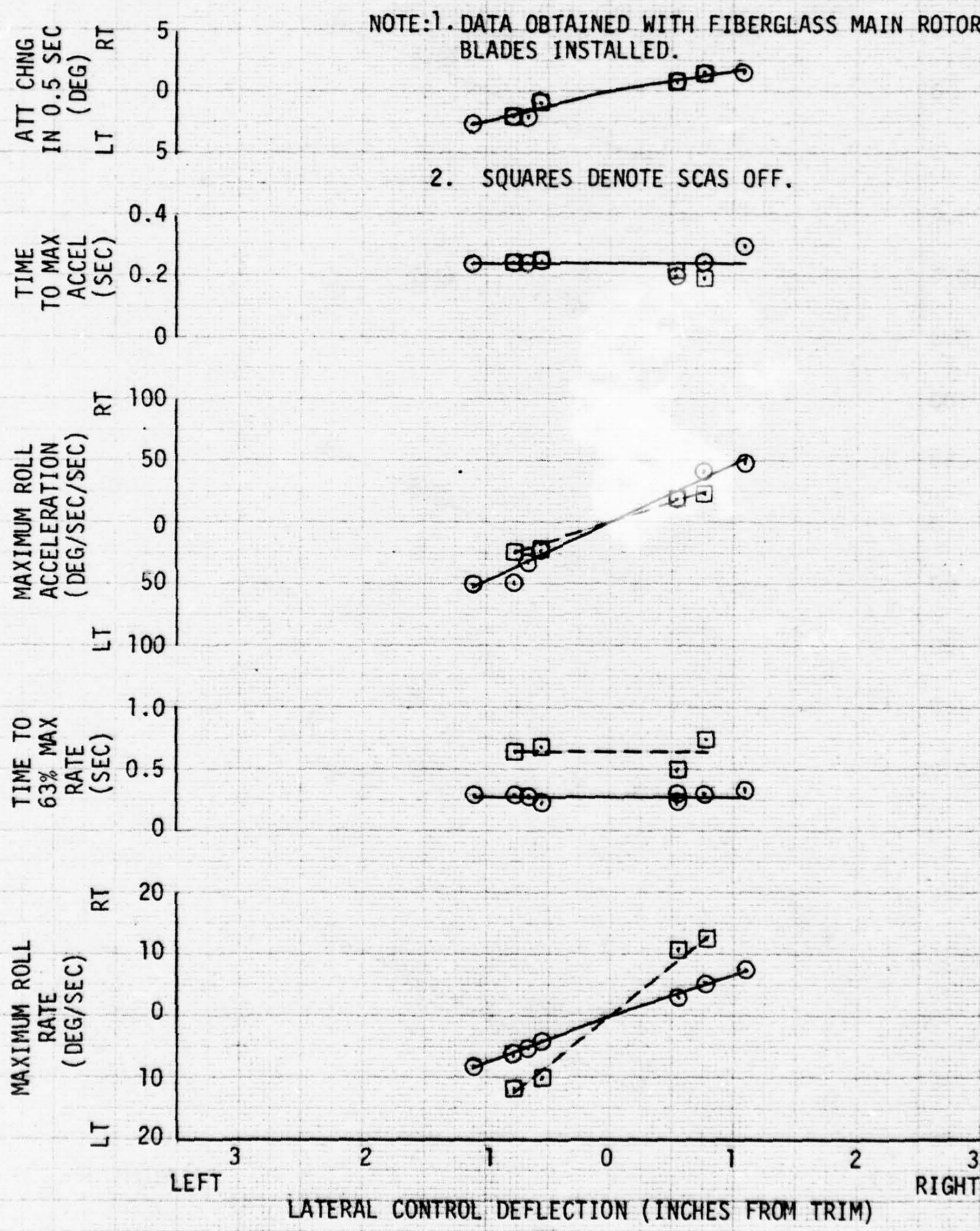


FIGURE 29
DIRECTIONAL CONTROLLABILITY
BELL MODEL 214
S/N 27001

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM CALIB. AIRSPEED (KT)
13340	139.9(AFT)	0.2LT(MID)	2040	26.5	303	0

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.

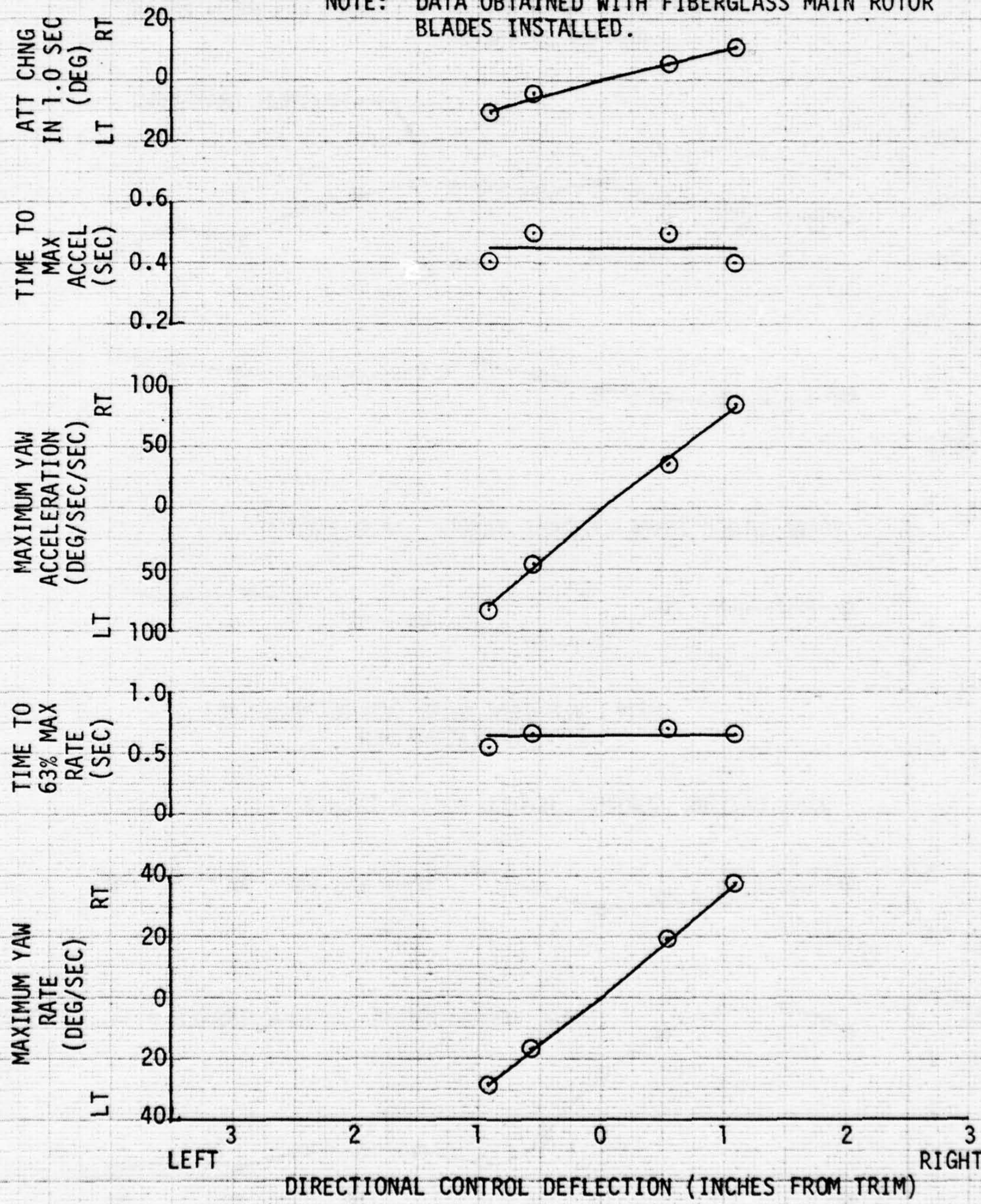


FIGURE 30
LOW-SPEED FORWARD AND REARWARD FLIGHT
BELL MODEL 214
S/N 27001

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	SKID HEIGHT (FT)
13640	136.2(FWD)	0.1LT(MID)	10120	1.0	300	10

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.

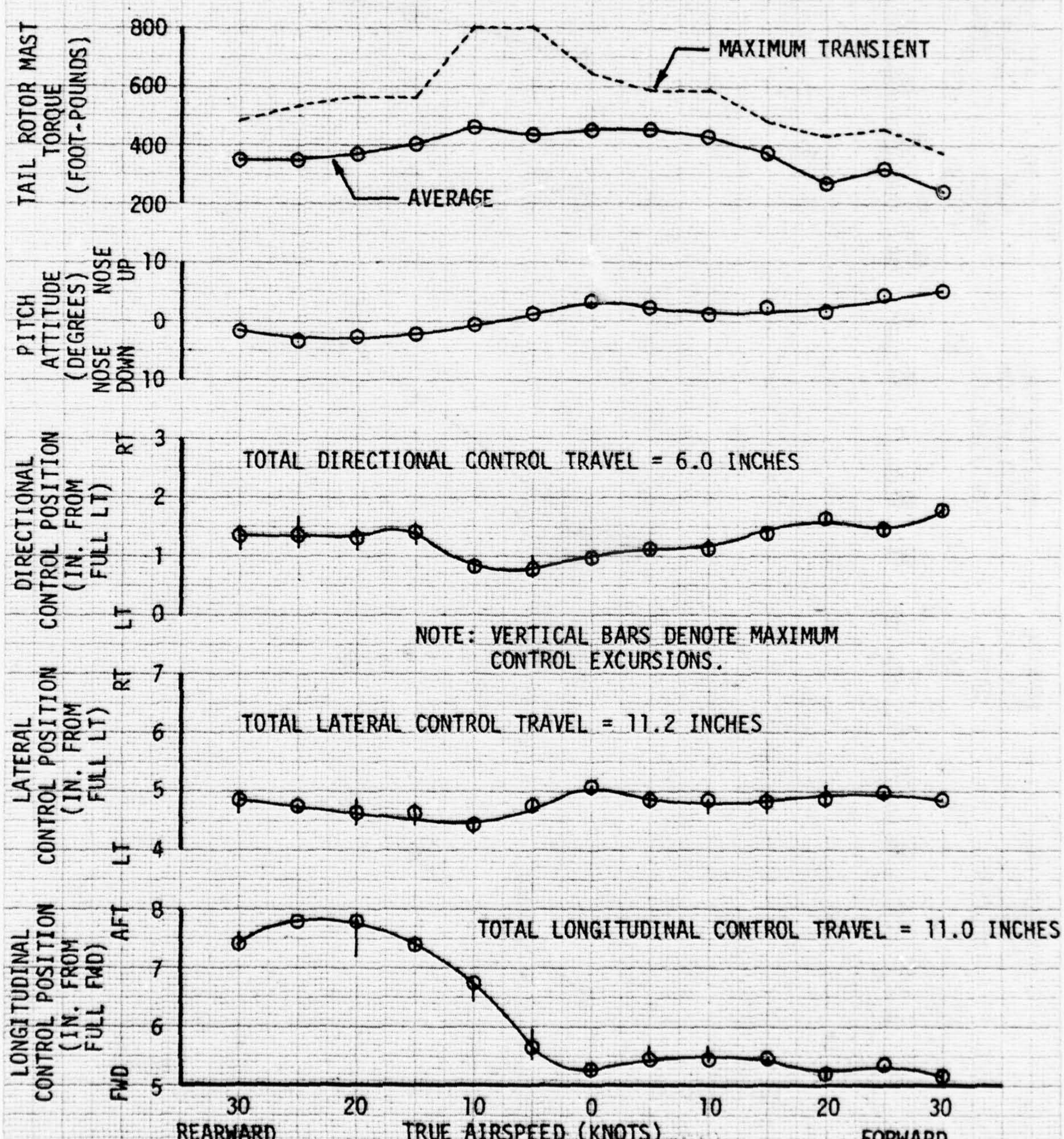


FIGURE 31
SIDEWARD FLIGHT
BELL MODEL 214
S/N 27001

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	SKID HEIGHT (FT)
13640	136.2(FWD)	0.1LT(MID)	10120	1.0	300	10

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.

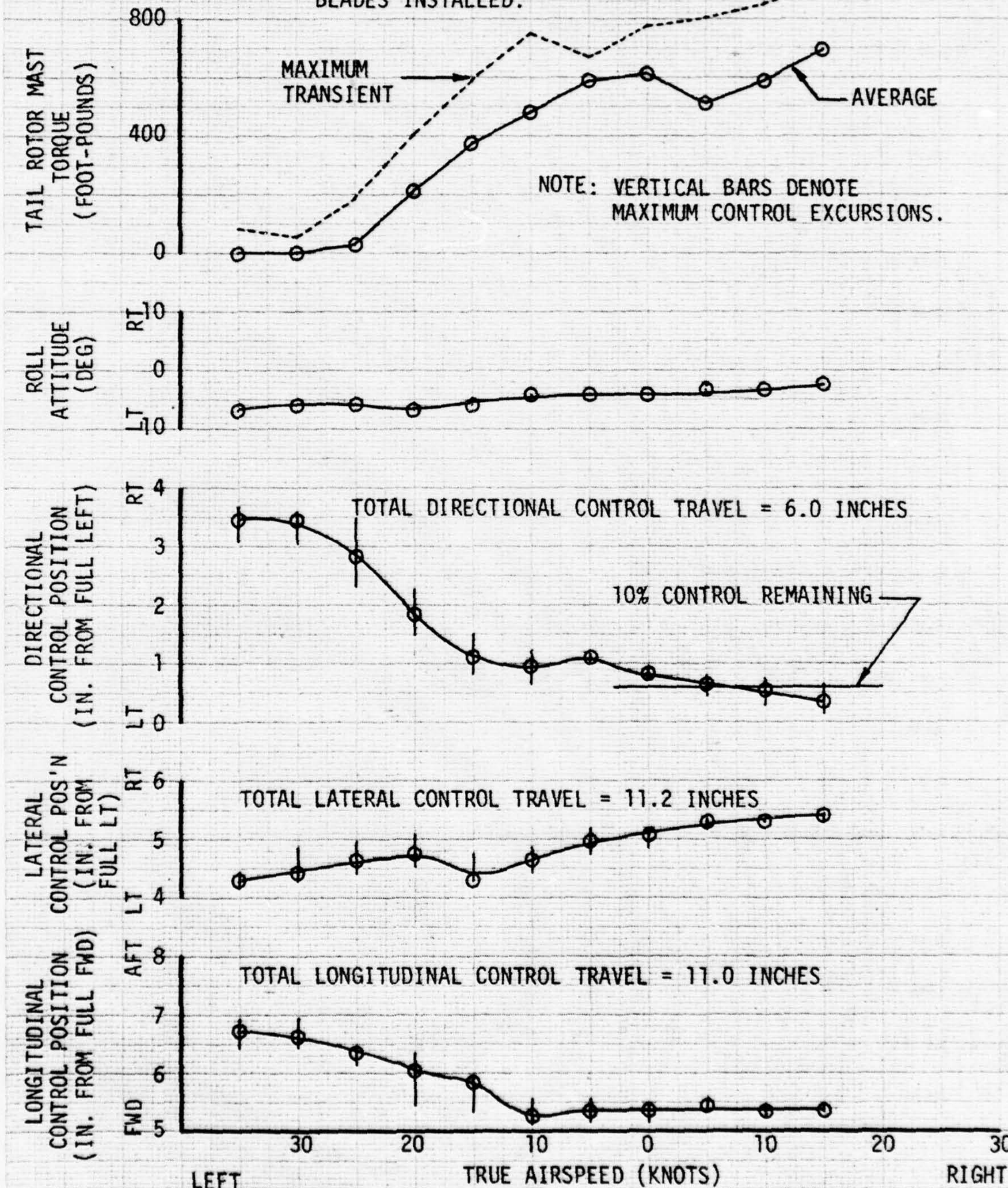


FIGURE 32
SIMULATED ENGINE FAILURE
BELL MODEL 214
S/N 27001

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	ENTRY CALIB A/S (KT)	ENTRY ENGINE TORQUE (%)	ENTRY FLIGHT CONDITION
9960	139.9(AFT)	0.2LT(MID)	6960	25	40	35	LEVEL

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.
TOTAL CONTROL TRAVEL: 11.0 INCHES LONGITUDINAL; 11.2 INCHES LATERAL; 6.0 INCHES DIRECTIONAL.

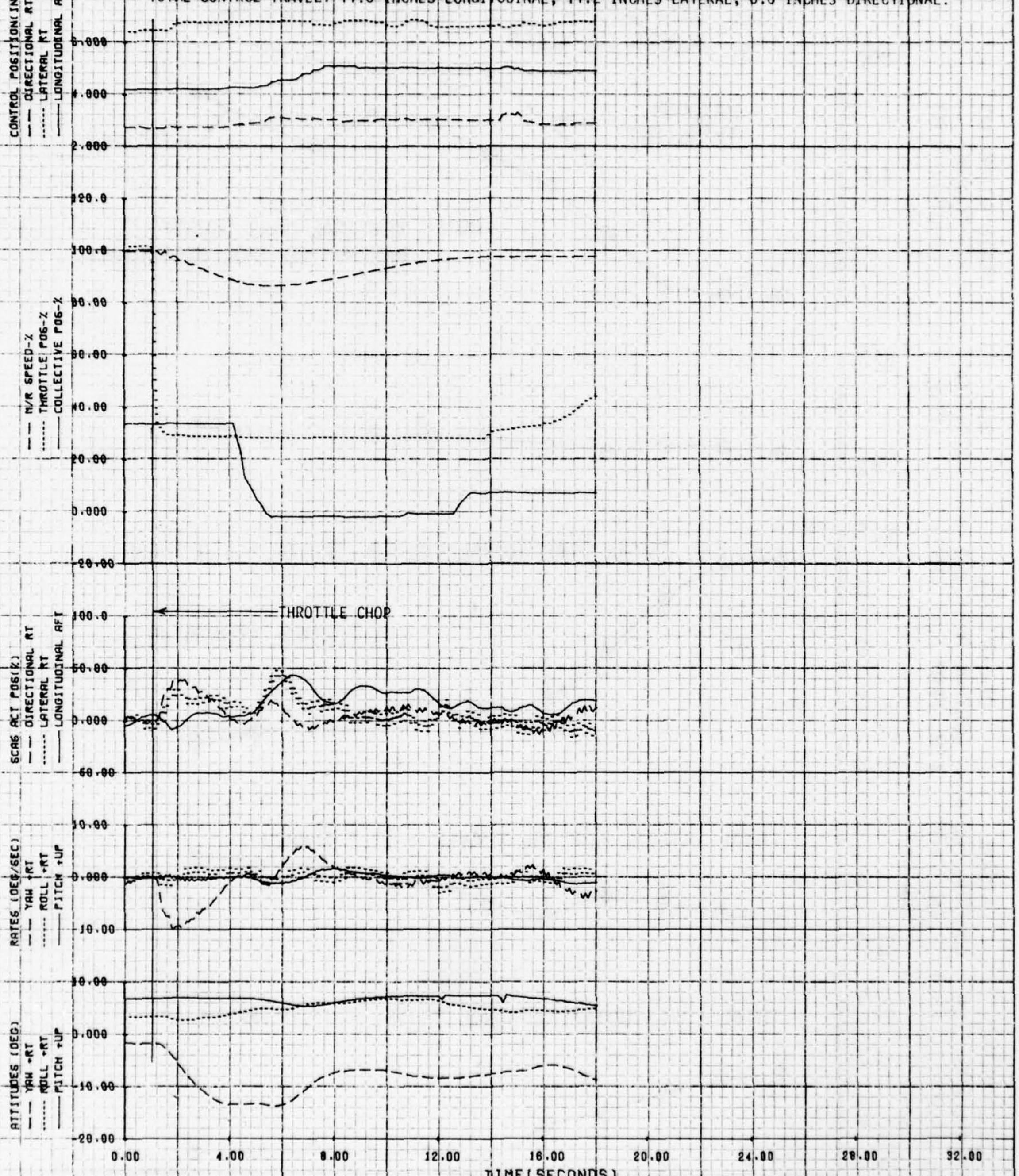


FIGURE 33
SIMULATED ENGINE FAILURE
BELL MODEL 214
S/N 27001

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	ENTRY CALIB A/S (KT)	ENTRY ENGINE TORQUE (%)	ENTRY FLIGHT CONDITION
9960	139.9(AFT)	0.2LT(MID)	6960	25	46	90	CLIMB

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.
TOTAL CONTROL TRAVEL: 11.0 INCHES LONGITUDINAL; 11.2 INCHES LATERAL; 6.0 INCHES DIRECTIONAL.

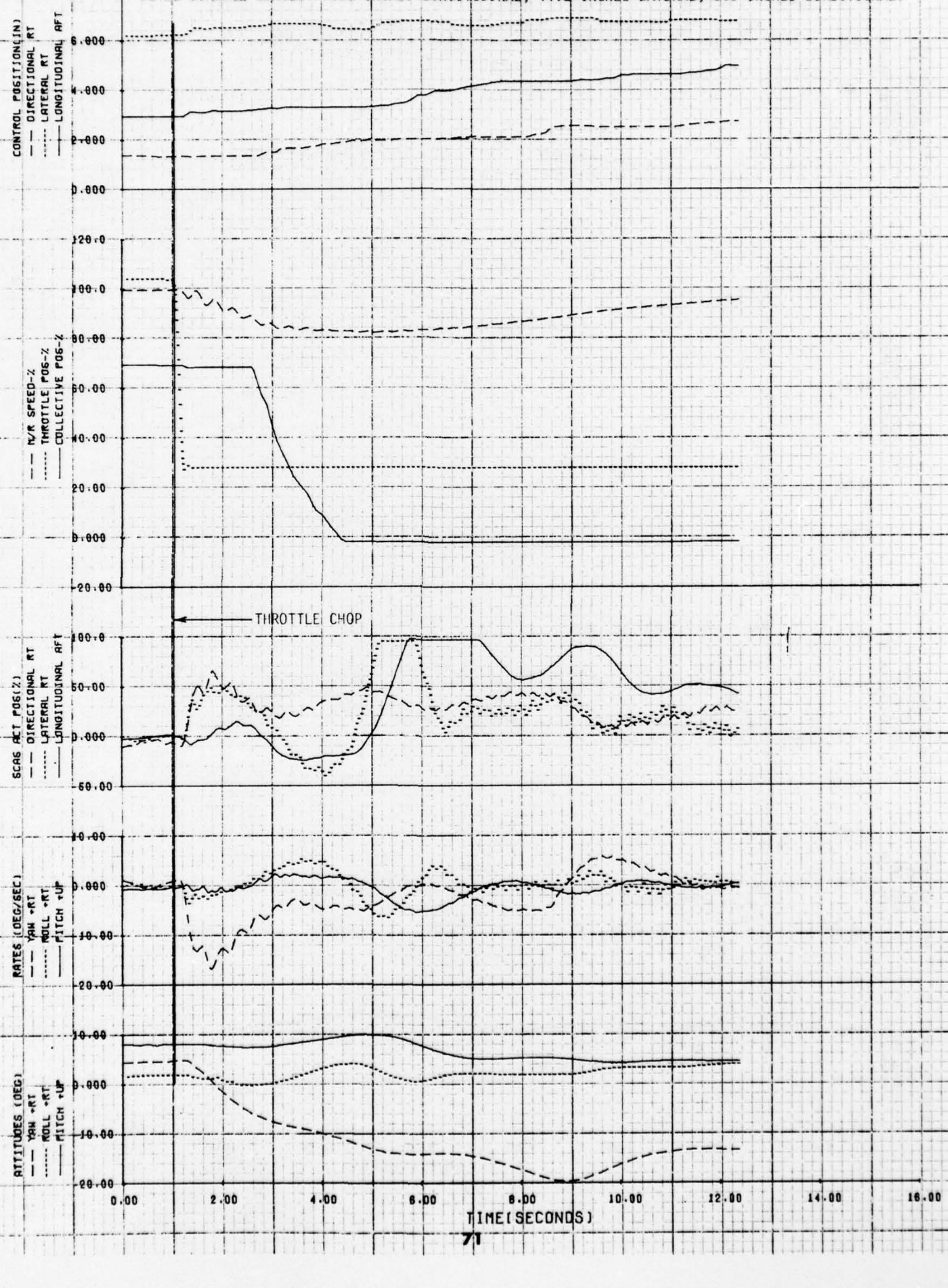


FIGURE 34
SIMULATED ENGINE FAILURE
BELL MODEL 214
S/N 27001

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	ENTRY CALIB A/S (KT)	ENTRY ENGINE TORQUE (%)	ENTRY FLIGHT CONDITION
9960	139.9(AFT)	0.2LT(MID)	6960	25	116	50	LEVEL

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.
TOTAL CONTROL TRAVEL: 11.0 INCHES LONGITUDINAL; 11.2 INCHES LATERAL; 6.0 INCHES DIRECTIONAL.

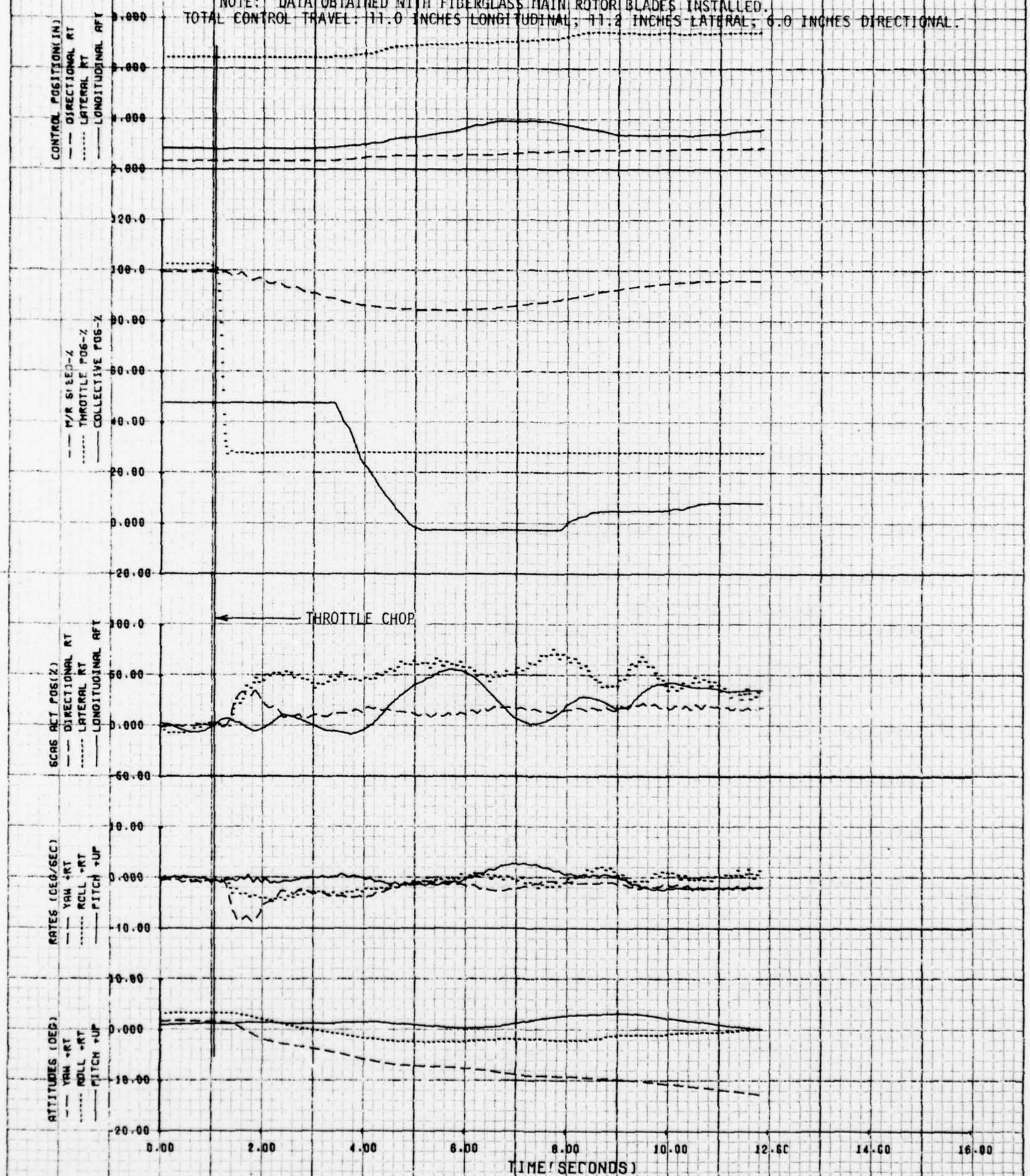


FIGURE 35
VIBRATION CHARACTERISTICS
BELL MODEL 214A
CG VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIG.	FLIGHT CONDITION
10480	135.7(FWD)	11660	7.0	299	.00523	CLEAN	LEVEL FLT

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.

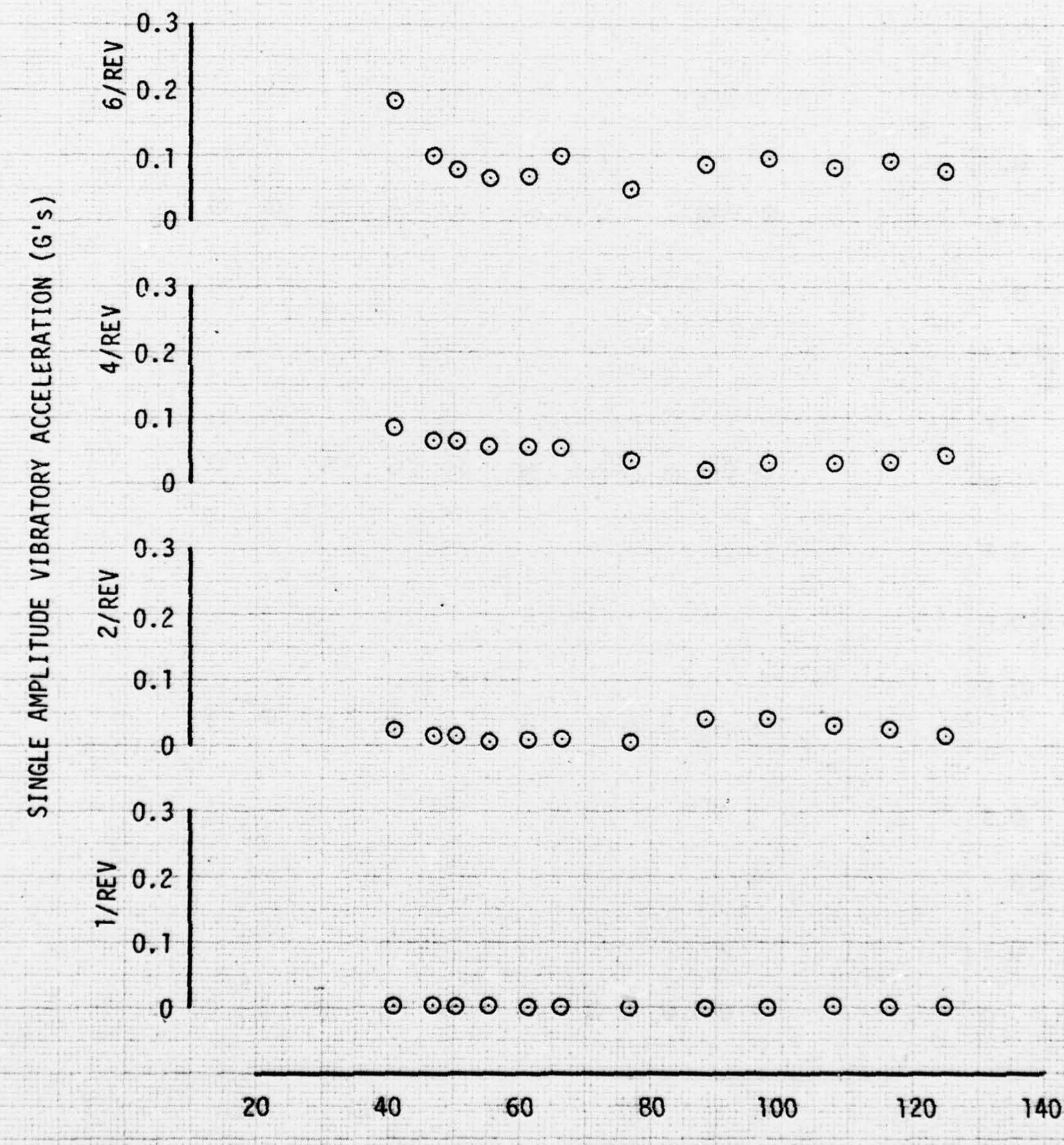


FIGURE 36
VIBRATION CHARACTERISTICS
BELL MODEL 214A
CG LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIG.	FLIGHT CONDITION
10480	135.7(FWD)	11660	7.0	299	.00523	CLEAN	LEVEL FLT

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.

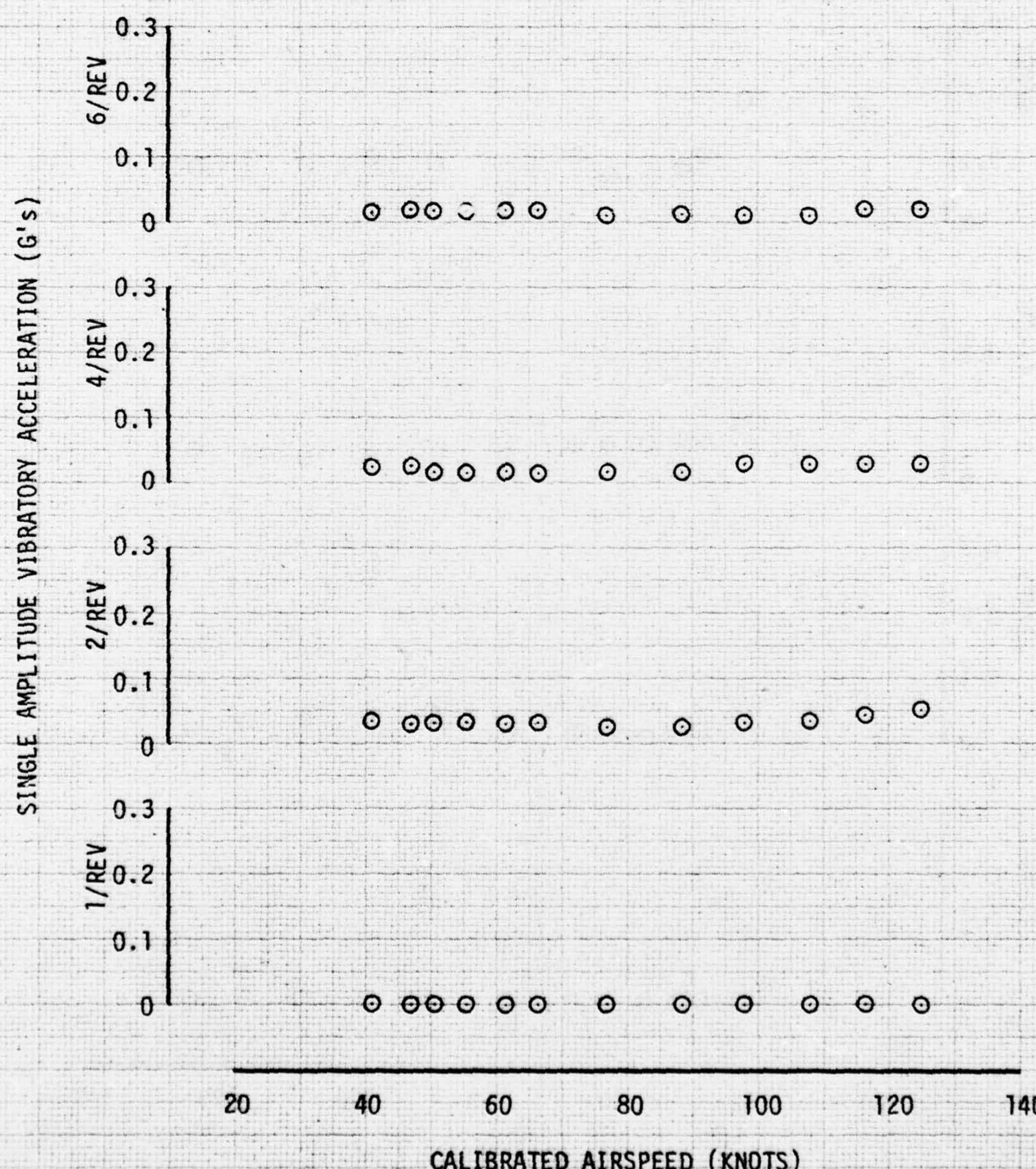


FIGURE 37
VIBRATION CHARACTERISTICS
BELL MODEL 214A
PILOT FLOOR VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIG.	FLIGHT CONDITION
10480	135.7(FWD)	11660	7.0	299	.00523	CLEAN	LEVEL FLT

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.

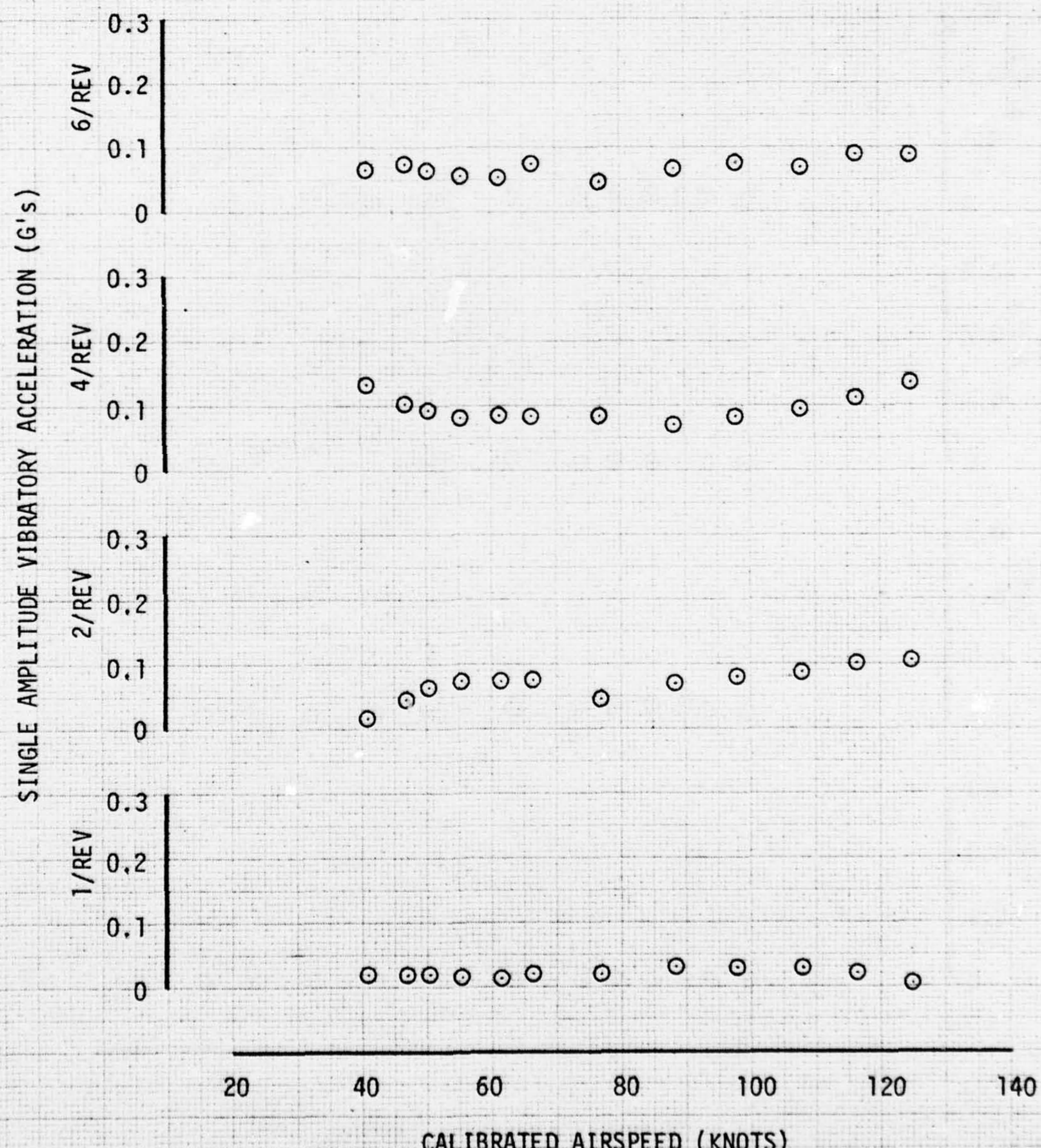


FIGURE 38
VIBRATION CHARACTERISTICS
BELL MODEL 214A
PILOT FLOOR LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIG.	FLIGHT CONDITION
10480	135.7(FWD)	11660	7.0	299	.00523	CLEAN	LEVEL FLT

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.

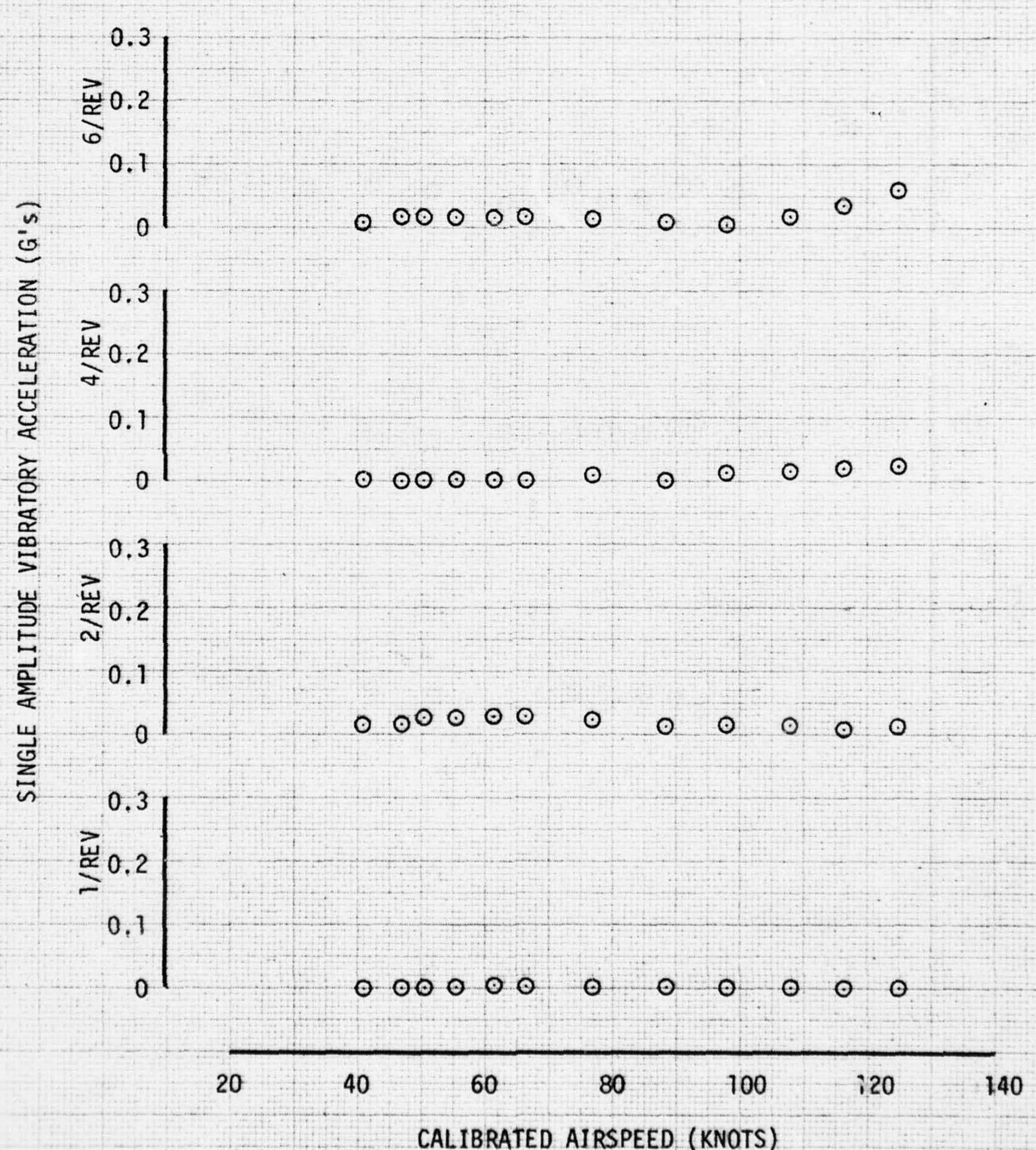


FIGURE 39
VIBRATION CHARACTERISTICS
BELL MODEL 214A
PILOT FLOOR LONGITUDINAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIG.	FLIGHT CONDITION
10480	135.7(FWD)	11660	7.0	299	.00523	CLEAN	LEVEL FLT

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.

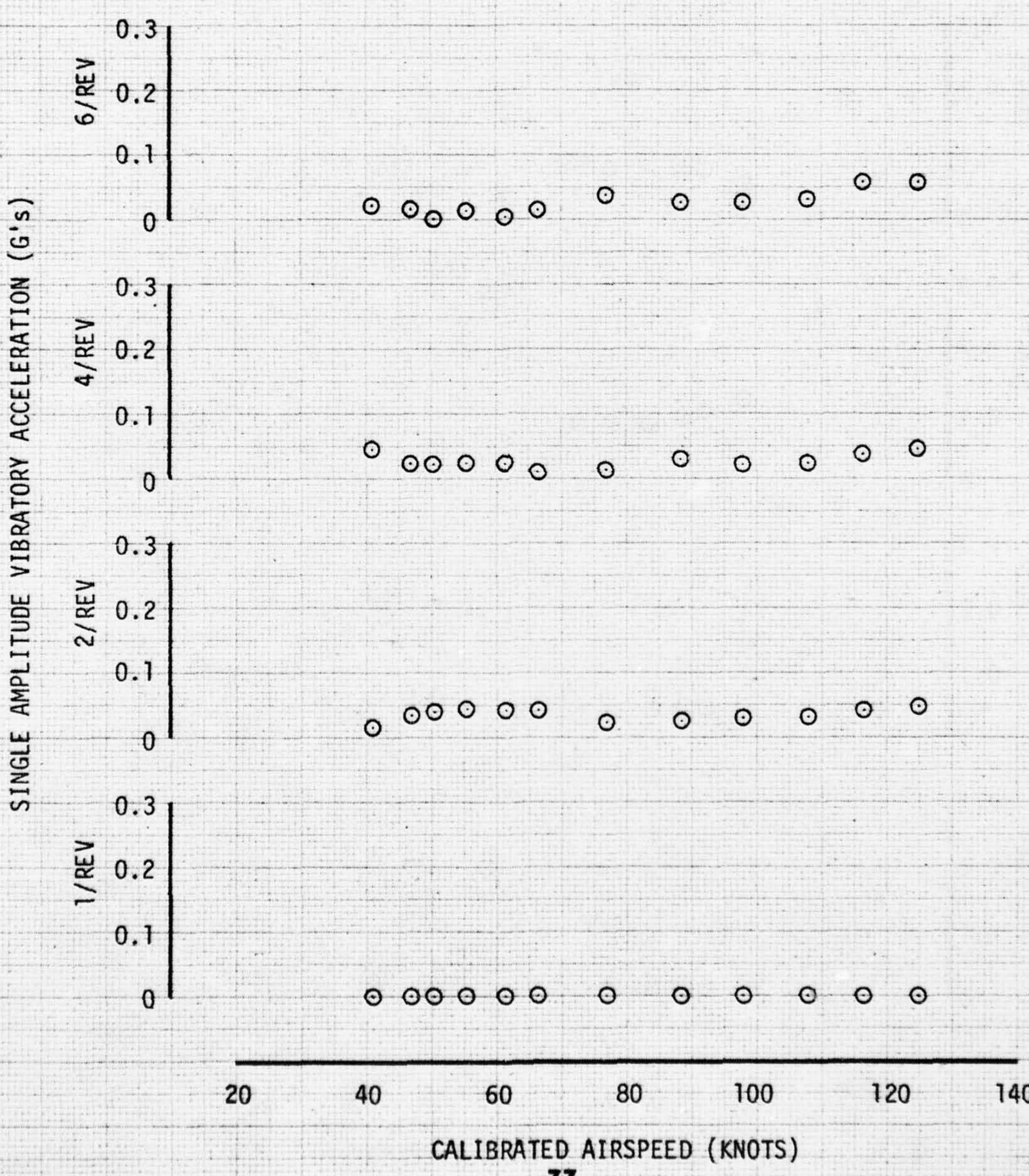


FIGURE 40
VIBRATION CHARACTERISTICS
BELL MODEL 214A
COPILOT FLOOR VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIG.	FLIGHT CONDITION
10480	135.7(FWD)	11660	7.0	299	.00523	CLEAN	LEVEL FLT

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.

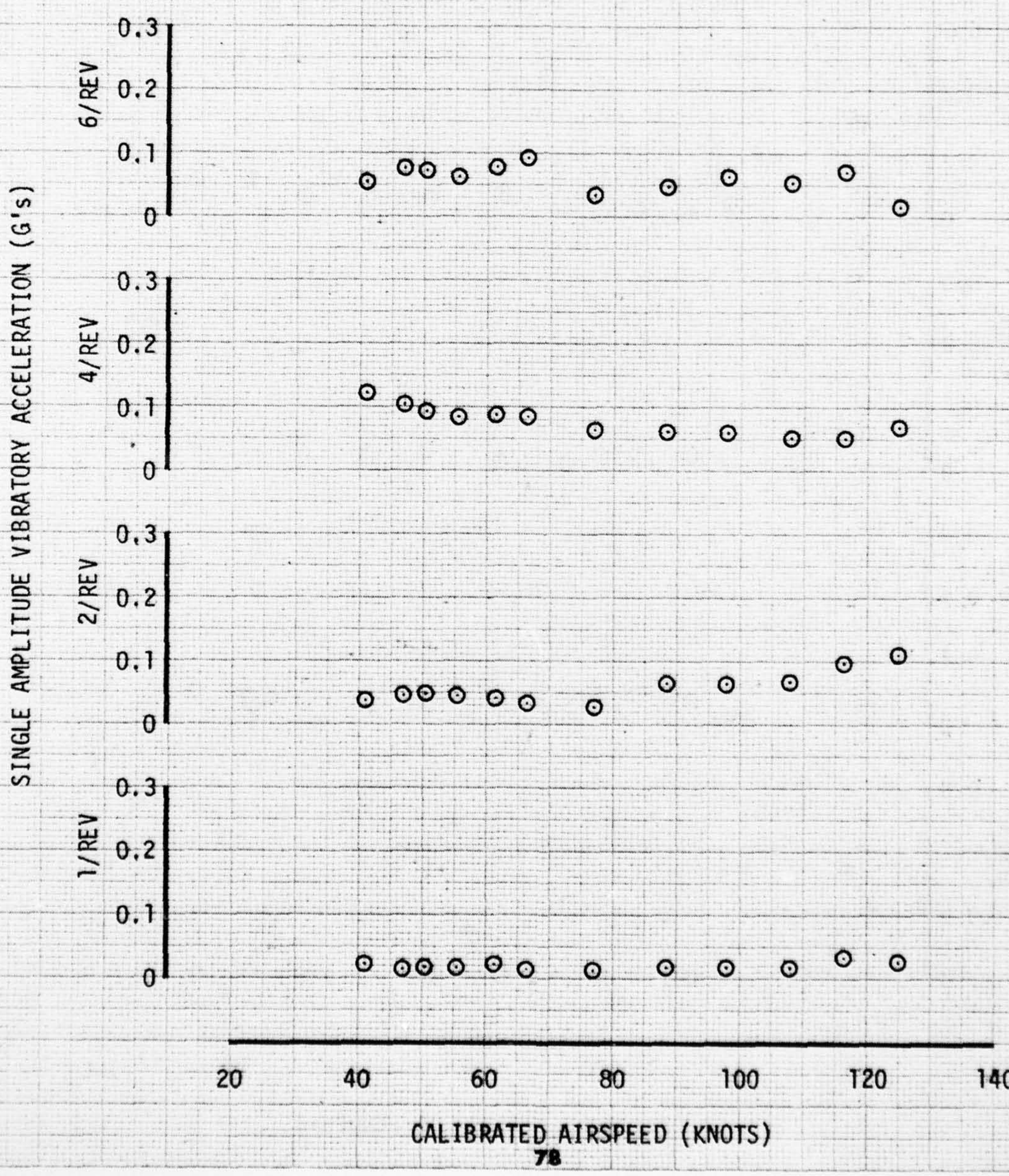


FIGURE 41
VIBRATION CHARACTERISTICS
BELL MODEL 214A
COPILOT FLOOR LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIG.	FLIGHT CONDITION
10480	135.7(FWD)	11660	7.0	299	.00523	CLEAN	LEVEL FLT

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.

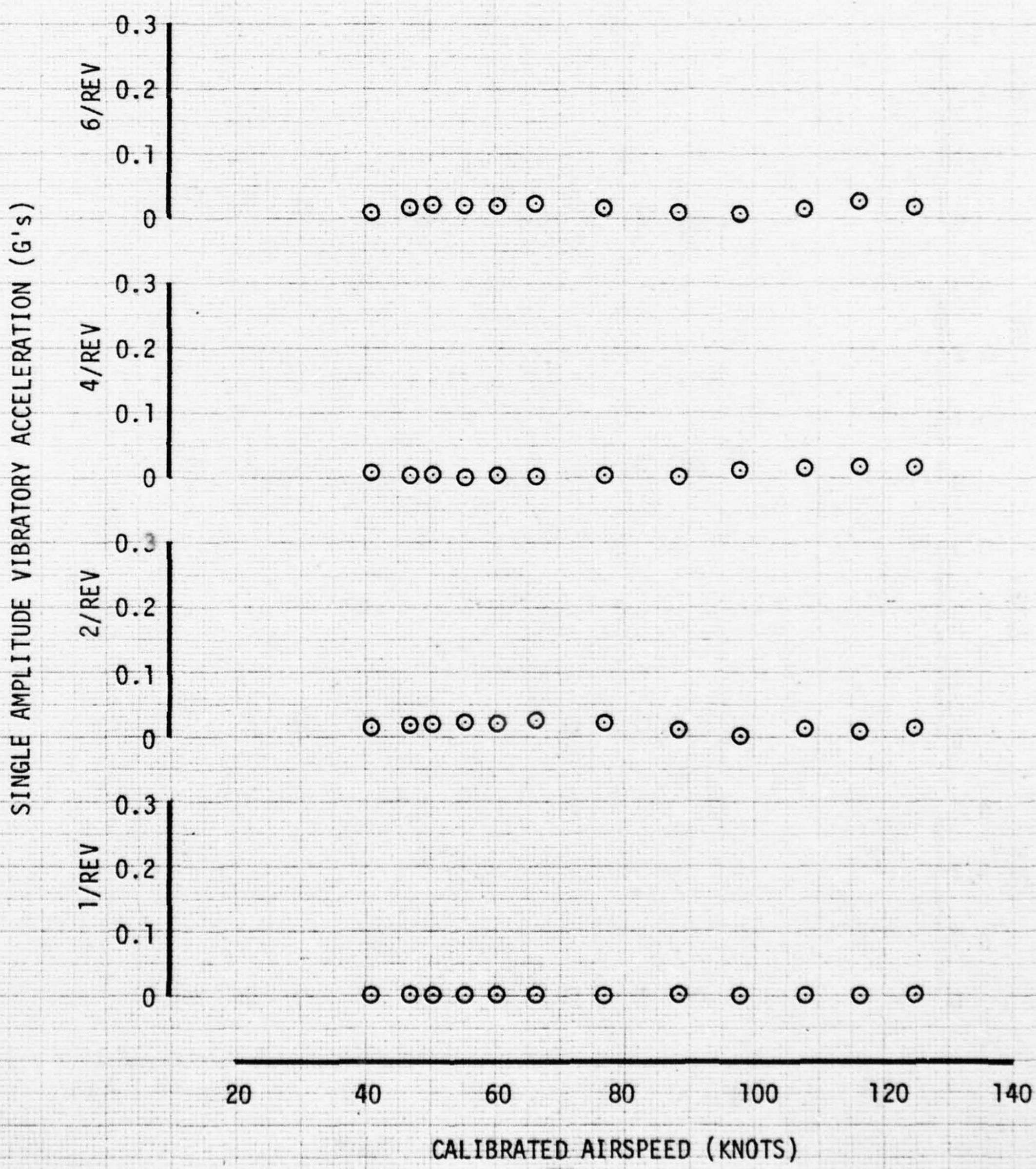
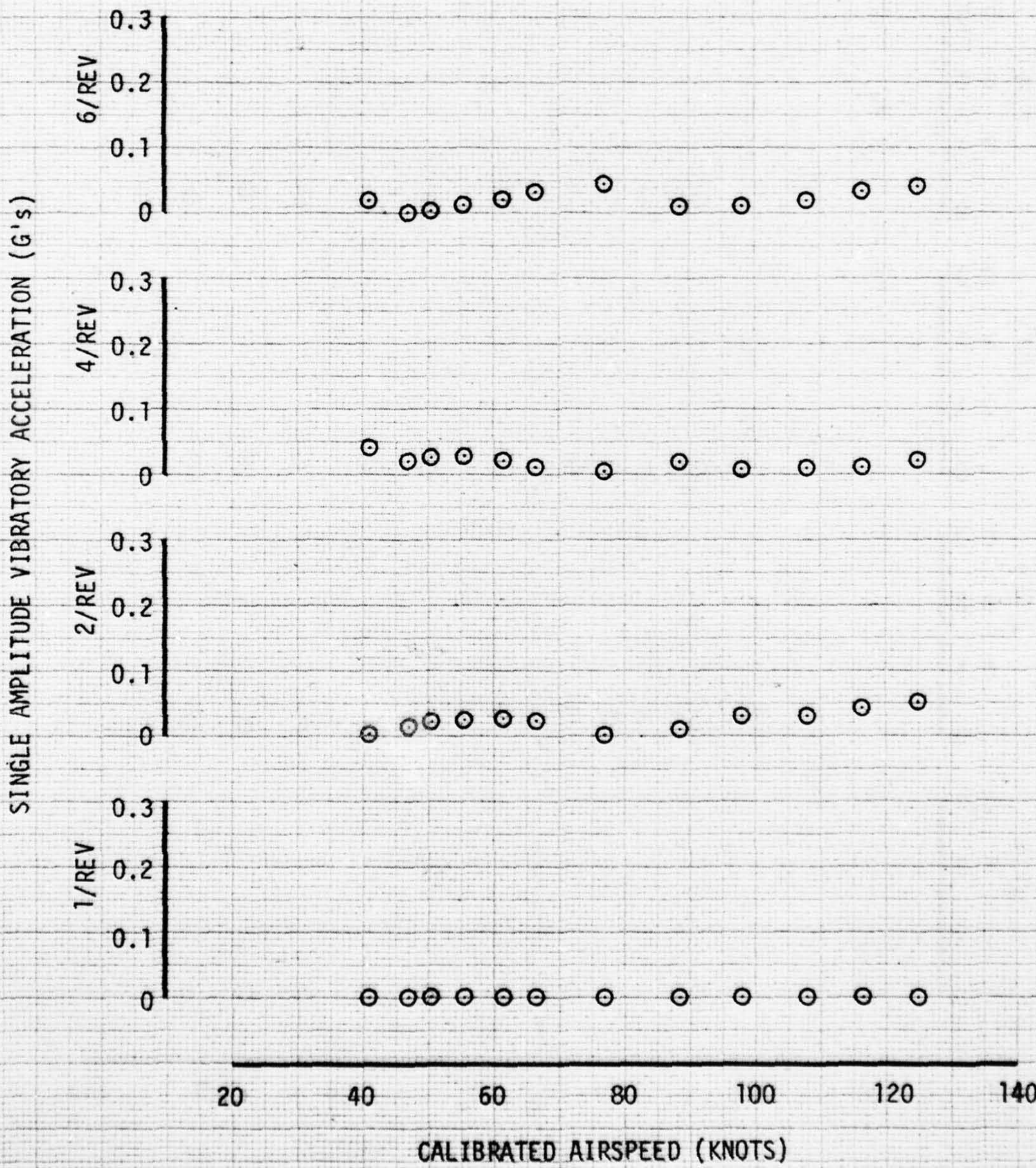


FIGURE 42
VIBRATION CHARACTERISTICS
BELL MODEL 214A
COPILOT FLOOR LONGITUDINAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIG.	FLIGHT CONDITION
10480	135.7(FWD)	11660	7.0	299	.00523	CLEAN	LEVEL FLT

NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.



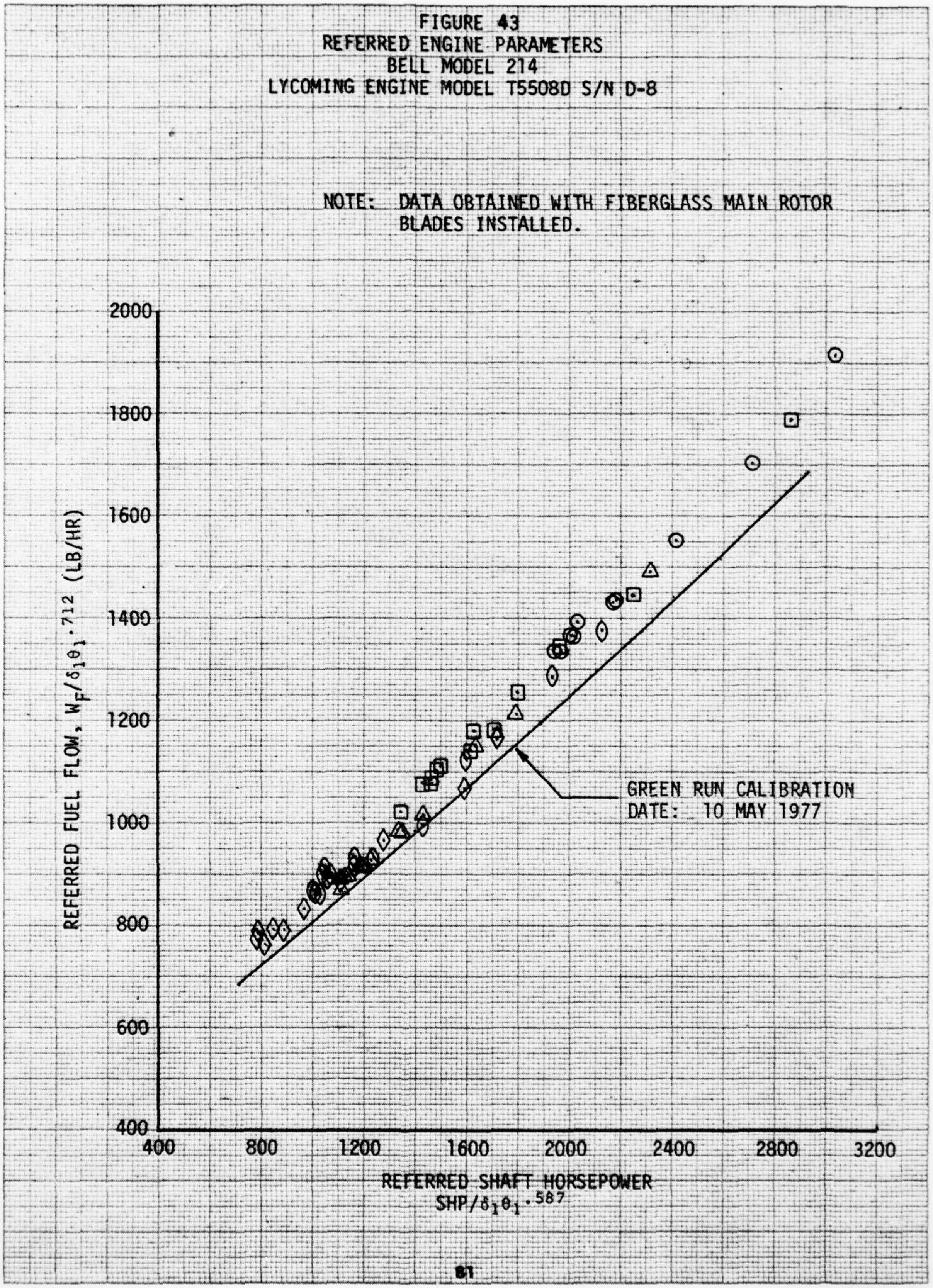
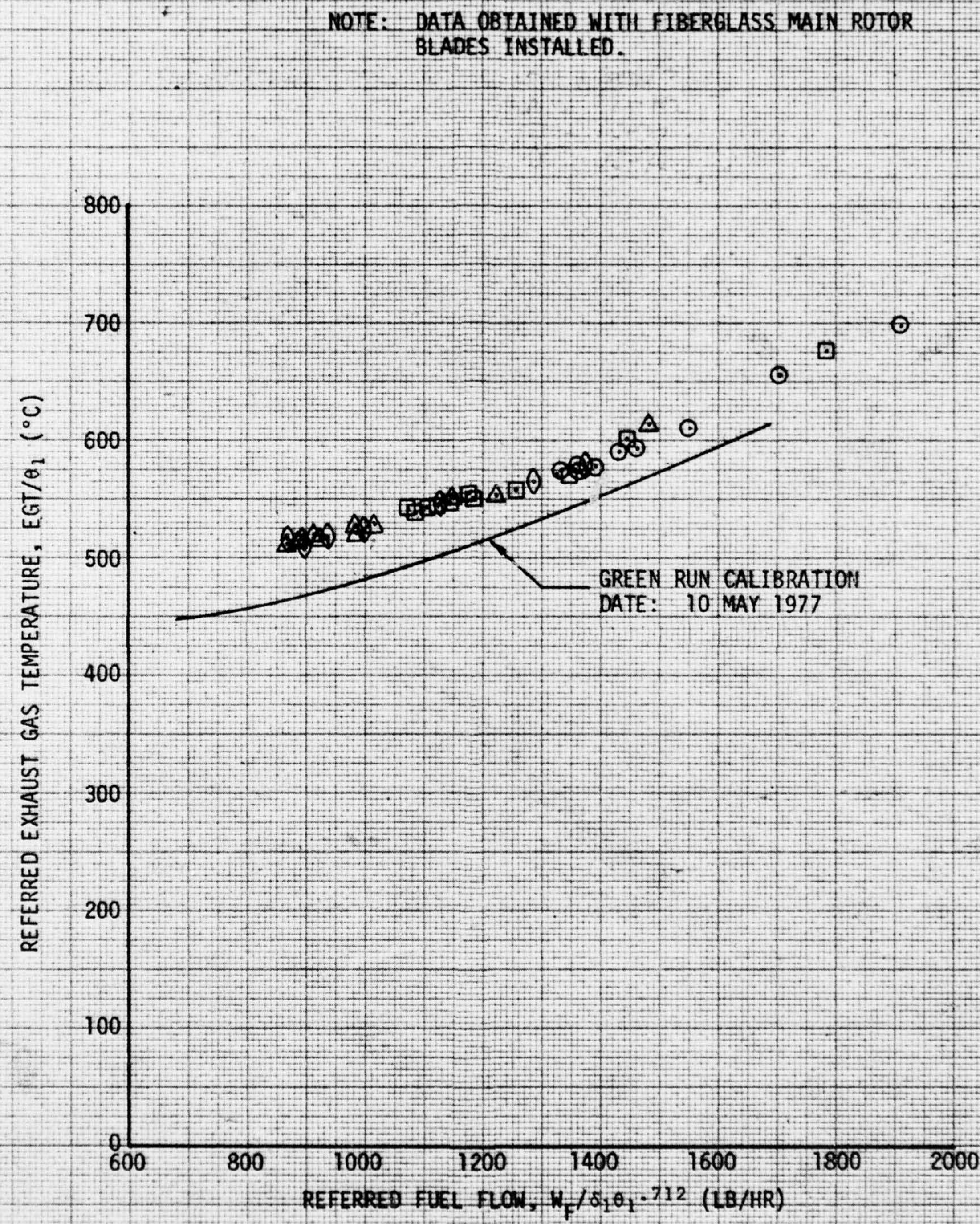


FIGURE 43
REFERRED ENGINE PARAMETERS
BELL MODEL 214
LYCOMING ENGINE MODEL T5508D S/N D-8
NOTE: DATA OBTAINED WITH FIBERGLASS MAIN ROTOR BLADES INSTALLED.
REFERRED FUEL FLOW, W_F/δ_1θ_1^.712 (LB/HR)
2000
1800
1600
1400
1200
1000
800
600
400
400
800
1200
1600
2000
2400
2800
3200
REFERRED SHAFT HORSEPOWER
SHP/δ_1θ_1^.587
GREEN RUN CALIBRATION
DATE: 10 MAY 1977

FIGURE 44
REFERRED ENGINE PARAMETERS
BELL MODEL 214
LYCOMING ENGINE MODEL T5508D S/N D-8



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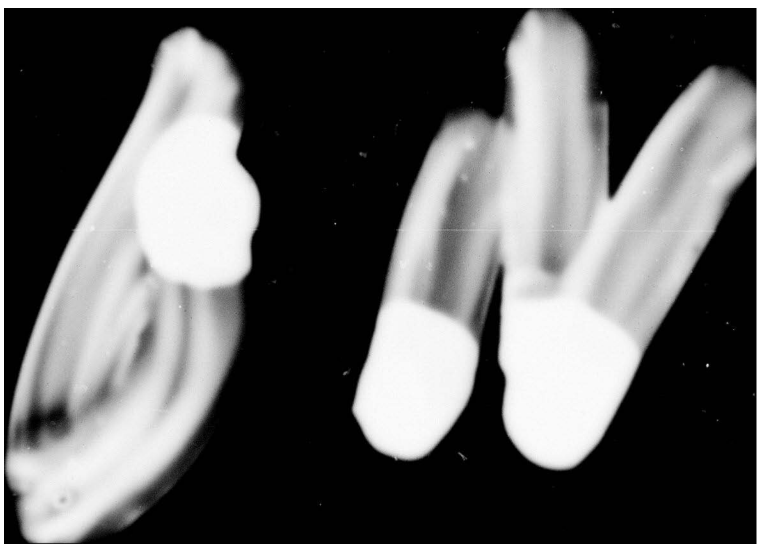

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LIMITED AIRWORTHINESS AND FLIGHT CHARACTERISTICS EVALUATION MOD--ETC(U)
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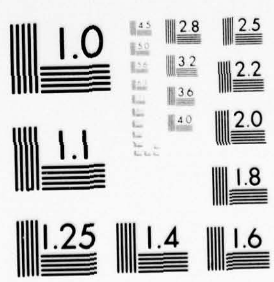

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USAAEFA Project No. 77-32

Final Report

Limited Airworthiness
and
Flight Characteristics Evaluation

Model 214A Helicopter
with Fiberglass Rotor Blades

United States Army Aviation Engineering Flight Activity
Edwards Air Force Base, California 93523

On Figure 17, Appendix E, the units of lateral control position should be "(IN. FROM FULL RIGHT)," and increasing left control position is toward the top of the page.